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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Technical Report No. 32-735

On the Evolution of Advanced Propulsion Systems for Spacecraft

Duane F. Dipprey
Jack H. Rupe
Richard N. Porter
David D. Evans

jpl

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CALIFORNIA INSTITUTE OF TECHNOLOGY
PASADENA, CALIFORNIA

July 15, 1965

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July 15, 1965

Prepared Under Contract No. NAS 7-100
National Aeronautics & Space Administration

Title:	On the evolution of advanced propulsion systems for spacecraft
Author(s):	Dipprey, D. F.; Evans, D. D.; Porter, R. N.; Rupe, J. H.
Abstract:	Use of earth-storable propellants in unmanned spacecraft propulsion systems for flights to near planets and moon
NASA Center:	Jet Propulsion Laboratory
Publication Date:	Jul 15, 1965
Document Source:	CASI
Online Source:	View PDF File
Document ID:	19650023544
Accession ID:	65N33145
Publication Information:	Number of Pages = 39
Report Number:	JPL-TR-32-735; NASA-CR-64609
Contract-Grant-Task Number:	NAS7-100
Price Code:	A03
Keywords:	EARTH (PLANET); MOON; PLANETS; PROPELLANTS; PROPULSION; PROPULSION SYSTEM CONFIGURATIONS; PROPULSION SYSTEM PERFORMANCE; SPACE FLIGHT; SPACECRAFT PROPULSION; STORABLE PROPELLANTS; UNMANNED SPACECRAFT; EARTH; MOON; PLANET; PROPULSION SYSTEM; SPACE FLIGHT; STORABLE PROPELLANT; UNMANNED SPACECRAFT;
Notes:	15 JUL. 1965 39 P REFS
Accessibility:	Unclassified; No Copyright; Unlimited; Publicly available;
Updated/Added to NTRS:	2006-11-06

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ABSTRACT

In a current research and advanced development program, the use of Earth-storable propellants in unmanned-spacecraft propulsion systems is under investigation. The term *Earth-storable* is applied to hypergolic propellants which exist in liquid form at $70 \pm 30°F$. Achievement of simplicity in propulsion-system design and operation is facilitated by the use of these propellants for flights to the near planets and the Moon. Propulsion-system requirements for such missions are briefly outlined, and the major hazards of space flight are discussed in relation to their effect on propulsion systems. The basic concepts of a proposed spacecraft propulsion system are presented, and component design is shown to be integrated with these concepts so that solutions to individual component problems are compatible. In the pursuit of system and component reliability, strong emphasis on simplicity and predictability is placed in opposition to the use of redundancy. The scope of this advanced development program has been delineated by three basic design choices, related to the use of (1) gas-pressurized tankage for propellant pumping, (2) flexible, impermeable barriers at the liquid–vapor interface in propellant tanks, and (3) thrust chambers of refractory and ablative materials which do not require regenerative cooling. For consistent achievement of stable, efficient combustion processes, compatible with their boundaries, the development of procedures for a priori injector design is shown to be mandatory.

Progress is summarized in the development of an engine, a gas generator, expulsion devices, regulators, and valves, all designed to meet the requirements of spacecraft propulsion systems burning Earth-storable propellants. A methodology for a priori injector design is presented, together with its application to a simple 10-element injector of the unlike-impinging-jet configuration. Nonreactive-spray properties are used to obtain quantitative descriptions of mass and mixture-ratio distributions, and methods of obtaining the mass-distribution model for an element are outlined. These techniques are being successfully applied to the evaluation of combustion-chamber materials. High-level, reproducible performance has been obtained in pyrolytic graphite chambers. Without film cooling, erosion rates are an order of magnitude lower than those commonly encountered.

On the Evolution of Advanced Propulsion Systems for Spacecraft[1]

I. THE USE OF EARTH-STORABLE PROPELLANTS IN SPACECRAFT PROPULSION SYSTEMS

Duane F. Dipprey

A. Introduction

At the Jet Propulsion Laboratory, interest in the propellants described as *Earth-storable* is related to their use in spacecraft designed for present and future unmanned exploration of the near planets and the Moon. One of the fundamental initial choices made in designing such a spacecraft is the temperature at which it is to operate. Given complete freedom in this choice, the designer would probably select the temperature of his own environment. Since, invariably, some portions of the spacecraft must be maintained at or near Earth temperatures, great simplification and ease of spacecraft design will be realized by keeping the entire spacecraft almost thermally homogeneous at these temperatures. Solar cells, batteries, and electronic equipment operate most effectively in this temperature range, and future manned missions will demand living compartments held at these temperatures. Certainly, the space-simulated ground testing of the spacecraft will be greatly facilitated if this testing can be accomplished at normal laboratory temperatures. Launch operations will be similarly simplified. There is yet another reason, perhaps more subjective but nonetheless important, for operating spacecraft at Earth temperatures whenever possible. The technology of equipment and materials has been built up naturally in the environment of Earth-based operation. Partially because of this, many common metals and plastics have been developed to exhibit their best combination of ductility and strength at these temperatures.

The choice of a temperature is not completely free, since it is constrained by the radiation-exchange properties of available spacecraft materials and by the part of the solar system in which the spacecraft is to operate. However, for operation in the general vicinity of the Earth's distance from the Sun, there are readily available structural materials and surface coatings for which the relation between the coefficients of absorptivity for solar radiation and emissivity for spacecraft radiation is such that the spacecraft can be maintained at Earth-surface temperatures. If the Earth itself is viewed, in a broad sense, as a spinning spacecraft, one might expect materials of Earth to have such properties. The variations in the solar constant, even as close to the Sun as Venus and as far from the Sun as Mars, still allow maintenance of spacecraft thermal equilibrium at near-Earth-surface temperatures with these same materials. That this is true for Venus missions has already been demonstrated by *Mariner II*. The midcourse propulsion system on this spacecraft exhibited temperatures varying from 80°F in the vicinity of Earth to 138°F in the vicinity of Venus.

[1]The material presented in this Report, in an abridged form, was published as three related articles in *Astronautics & Aeronautics* for June 1965.

Mariner IV, now on its way to Mars, will provide a similar demonstration for flights to that planet. Propulsion-system temperatures when *Mariner IV* was near the Earth were approximately 71°F, and predicted temperatures near Mars encounter are around 50°F.

Since the maintenance of spacecraft at near-Earth temperatures is both desirable and possible, propellants which can be stored and used at these temperatures become important candidates for use in on-board propulsion systems. This has led to the categorization of certain propellants as *Earth-storable*. We have used this term with reference to those rocket propellants which exist in liquid phase at temperatures commonly encountered on the Earth's surface, say 70 ± 30°F. Propellants thus designated necessarily have vapor pressures below some reasonable operating tank pressure, say 100 psia, in the prescribed temperature range. It is also arbitrarily implied that the bipropellants in this class are hypergolic, meaning that the propellants ignite on contact, thus negating the need for an ignition system. Some of the propellants which we classify as Earth-storables are shown in Table 1, together with some of their temperature-dependent properties. It may be noted that their characteristics are not greatly different from those of water.

Both *Mariner II* and *Mariner IV* used a storable propellant (monopropellant hydrazine) in a small propulsion system which provided for course-correction maneuvers. In these spacecraft, the propellant mass was less than 10% of the total spacecraft mass. However, for future missions, the spacecraft mass may be more than half propellant; thus, the requirements for spacecraft temperature control will be increasingly dependent on the thermal characteristics of the propellants. Hence, there exist compelling reasons for expanding storable-propellant technology, both by learning to use more effectively those propellants that are now available and by developing new propellants in this class.

B. Research and Advanced Development in Storable-Propellant Technology

The general goal of any advanced development and applied research program must necessarily be to create and perfect the new techniques and technology needed to meet foreseen requirements. Propellants that might be classed as Earth-storable have been in use throughout the history of liquid-propellant rocketry, but their use in spacecraft propulsion systems requires an ascendant emphasis on reliability and a good understanding of the special conditions imposed by the space environment. Exceptional component reliability and operational predictability must be achieved. Future spacecraft may have as many as 50,000 identifiable electronic and active mechanical parts, any one of which could cause partial or total failure of a mission. Hence, the degree of unreliability attributable to any single failure mode or subsystem becomes immeasurably small. High performance (low system mass) and low unit cost will be of lower-order importance than reliability in many future spacecraft-propulsion applications. As the new larger boost vehicles become available, the great emphasis that has been put on low subsystem mass will be shifted to favor increased reliability for the longer-duration and more complicated

Table 1. Temperature-related properties of selected Earth-storable propellants

Propellant	Density at 70°F, g/cc	Boiling point, °F	Freezing point, °F	Vapor pressure at 100°F, psia	Viscosity at 40°F, cp
Fuels					
Hydrazine (N_2H_4)	1.01	236	35	0.6	1.22
Unsymmetrical dimethylhydrazine (UDMH)	0.79	146	−71	5.3	0.72
50% N_2H_4/50% UDMH	0.90	158	20	4.6	1.16
Monomethylhydrazine (MMH)	0.88	193	−63	1.9	1.18
Water (H_2O)	1.00	212	32	0.9	1.53
Oxidizers					
Nitrogen tetroxide (N_2O_4)	1.44	70	12	30.7	0.50
90% N_2O_4/10% nitric oxide (NO)	1.41	10	−10	74.0	0.36
Red fuming nitric acid (RFNA)	1.57	136	−68	4.9	1.79
Hydrogen peroxide (H_2O_2)	1.45	302	31	0.1	1.74
Chlorine trifluoride (ClF_3)	1.82	53	−107	39.5	0.51

missions. Related to cost is the fact that, for many missions, only 2 or 3 flight units and perhaps as many as 10 spares and auxiliary test units will be required over a period of years. Thus, the high costs incurred in attaining reliability by the production of individual components with tight tolerance control, by complete inspection, and by elaborate testing will actually represent an economy when viewed in reference to the success of the mission.

A primary goal of our research and advanced development of spacecraft propulsion systems is to achieve the required reliability through simplicity, predictability, and a thorough understanding of the system and its components. First emphasis is on simplicity of design—minimum number of components, minimum number of seals, simple operating sequences, and simple interfaces with other subsystems. The use of storable hypergolic propellants is compatible with this goal.

The strong emphasis on simplicity and predictability is placed in opposition to the use of redundancy in pursuit of high reliability. This is done advisedly. The acceptable component unreliabilities for spacecraft propulsion systems are so small, and the number of units to be put into service is so small, that direct measurement of the unreliability by testing is out of the question. Instead, extremes in tolerance buildup and extremes in environmental test conditions are employed to ferret out failure modes. Once a failure mode is discovered, the advanced development effort is then to be concentrated on redesign to correct the incipient weakness, rather than resorting to redundancy to minimize its effects. One of the principal difficulties with the use of redundancy under conditions of very small acceptable unreliability is assessment of the loss in reliability associated with the use of a redundant component. It is difficult to establish whether or not the added assembly operations, the need for increased testing, and the additional possibilities of leaks and contamination introduced with a redundant component have introduced a net unreliability.

C. Injector Design Precepts

Having adopted system concepts as simple as is practicable, we have directed our attention in the research and advanced development of storable-propellant systems toward complete understanding and control of the components and subsystems. The least understood, and often least predictable, part of a liquid-propulsion system is the rocket engine itself. Since complete interpretation of rocket combustion processes is an elusive and perhaps unattainable goal, our efforts are concentrated on the lesser goals of phenomenological understanding and predictability. It is taken as a first principle that, if injector hydraulics are well controlled, the specific impulse, heat-rejection distribution, and combustion stability will follow as a direct and reproducible consequence of the particular injection geometry being investigated.

To control injector hydraulics, extensive use is being made of injection elements having large length-to-diameter ratios. These elements yield dynamically and kinematically stable streams with well established velocity distributions. Bistable or multistable flow patterns, resulting from short injection orifices operated with significantly large manifold cross velocities, are thus avoided, and performance and heat rejection for a given design become extremely steady and reproducible. Variations of design injection patterns can then be carried out systematically to yield uniformly low heat rejection with minimum or no sacrifice in performance, and also to yield well localized reaction zones in which resonant combustion can be inhibited by means of cooled baffles. Through the use of these concepts, considerable success in isolating the key factors in successful injector design for storable propellants has already been obtained. Some aspects of our work in this field are described and discussed in Section II of this Report.

D. Mission Requirements for the Earth-Storables

The requirements for storable-propellant spacecraft propulsion systems are perhaps best introduced in terms of a specific mission. One mission which places extensive demands on spacecraft propulsion is the Mars orbiter, which has been and continues to be the subject of much interest and study (Refs. 1 and 2). Consideration has been given to the use of a *Saturn I-B* boost vehicle which would launch a spacecraft weighing in excess of 5000 lbm on a Mars trajectory. An artist's conception of such a spacecraft is depicted in Fig. 1. The principal propulsive maneuver of this mission will be performed by a retro-rocket which will inject a portion of the spacecraft into an orbit about Mars. One proposed scheme calls for a low-thrust, single-thrust-level, restartable propulsion system. This unit will be operated for short periods several times during transit to Mars, in order to provide spacecraft course corrections; it will then be operated for an extended period to provide the velocity decrement required to achieve Mars orbit; and, finally, it may be

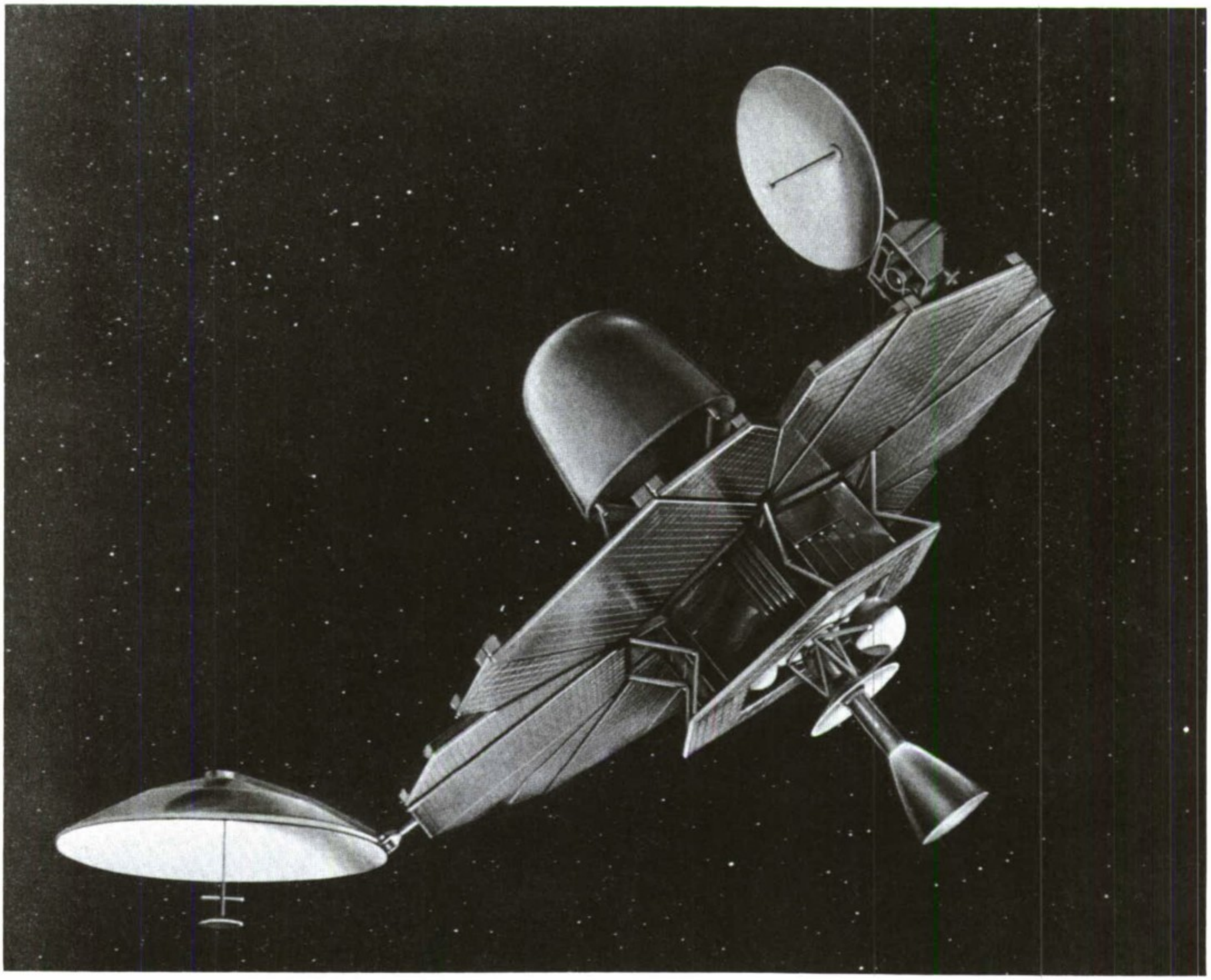


Fig. 1. Artist's conception of a Mars orbiter-lander spacecraft

started once or twice more to provide orbit modifications for later phases of the mission. For the 5000-lb spacecraft and a delivered impulse of about 1,000,000 lbf-sec, the appropriate engine thrust level is about 1000 lbf. This implies a total burn time of the order of 1000 sec.

The hypergolicity of the Earth-storable propellants and their compatibility with the spacecraft thermal environment make them particularly well suited to this mission. Hypergolicity, with the attendant simple start sequences, is very important in a system that may be started as many as five times in a single flight. The most interesting cryogenic competitors to the Earth-storable bipropellants for the Mars orbiter mission are fluorine-hydrazine and oxygen difluoride–diborane, either of which will most certainly be considered for uprating of later Mars orbiters. Either combination could yield 10 to 20% payload-in-orbit advantage over the Earth-storables, since neither exhibits the very low density and very low storage temperature which make hydrogen combinations noncompetitive for this mission.

The major advantage, other than storage temperature, of the Earth-storables over the competing cryogenic propellant combinations is lower flame temperature. As discussed subsequently, the desire for the ultimate simplification of spacecraft propulsion systems calls for non-regeneratively cooled thrust chambers, either ablative or refractory. Discovery of chamber materials that will contain the fluorine–hydrazine or oxygen difluoride–diborane reaction for 1000 sec without regenerative cooling is still an unrealized objective, even though the search is being actively pursued (Ref. 3). The combustion temperatures of the storable bipropellants are of the order of 2000°F lower than those of the higher-performance propellants. Materials, both ablative and refractory, are now being tested which have the potential of meeting the long-burning-time requirements with the Earth-storables.

E. Propulsion-System Survival in the Space Environment

Transcending all other mission demands is the requirement that the propulsion system survive unattended and function repeatedly for year-long periods in the space environment. The major hazards and design constraints imposed by the environment are thermal radiation, ionizing radiation, vacuum, weightlessness, and micrometeoroid flux. The compatibility of the storables with the attainable thermal environment is discussed above.

1. Ionizing Radiation

One of the hazards of ionizing radiation in space is the possible decomposition of propellants. Estimates of the radiation dosage rates (Ref. 4) show the Earth's radiation belts to be the primary source. From dosage-rate estimates and typical trajectories, an estimate can be made of the accumulated radiation dosage of a Mars-bound spacecraft. This is several orders of magnitude less than that required to degrade liquid propellants (Ref. 5). Should Mars also prove to have radiation belts, this conclusion must be reexamined for the Mars orbit chosen, but the possibility of a serious problem arising from ionizing radiation appears to be very remote.

2. Vacuum

The hard vacuum of space poses materials problems that could affect propulsion-system designs. Thermal sublimation of materials exposed to vacuum for a year is of little concern with regard to the metals and the inorganic coatings commonly used on spacecraft; however, the use of organic polymers is another matter. Plastic valve seats exposed on one side to hard vacuum may volatilize material that is then recondensed on other surfaces in such a way as to interfere with valve action. If metal-to-metal seats are used, the possibility of cold-welding of the mating parts exists when the films of atmospheric gases and oxidation are worn off and are not replenished. (Valve designs compatible with hard vacuum are discussed in Section III of this Report.) A more serious problem involving vacuum sublimation could occur in ablative rocket chambers. The integrity of the ablating surfaces is maintained by the sublimation of volatile components from the surface into the hot-gas stream during the motor burn. At shut-off, the surfaces continue to sublime while remaining momentarily at temperatures ranging downward from 5000°F. The released gases could recondense on the injector, fouling it for subsequent operations.

3. Weightlessness

Although not strictly a condition of the space environment, weightlessness is a condition of space flight and must be dealt with in all propulsion systems requiring start-up with liquid-phase propellants. Under zero-g conditions, the density difference between the liquid and vapor phases in the propellant tanks no longer serves to orient the interface. When perturbing forces on the spacecraft are completely negligible, surface-tension forces will localize the liquid by forming a surface of minimum surface energy, but this effect cannot always be depended upon for the needed fluid control in spacecraft propulsive operations. Often, just prior to engine start-up, the spacecraft will have undergone an attitude-change maneuver to align the main rocket to a predetermined orientation. This maneuver, usually performed with very small propulsive jets, will probably upset the delicate balance of the fluid-surface configuration established by surface tension. Surface control can be accomplished by constructing a movable barrier between the liquid and vapor phases. Among the possible barriers are a flexible bag, as used in the *Mariner* and *Ranger* spacecraft systems, and a flexible diaphragm attached to the periphery of the tank.

4. Micrometeoroid Flux

The micrometeoroid flux may prove to be the most serious hazard of the space environment. Propellant tanks on the Mars orbiter will expose large surface areas to possible meteoroid puncture, and any puncture is almost certain to cause a mission failure. According to the penetration laws summarized by Whipple (Ref. 6), the mass of meteoroids that can puncture aluminum tanks (of a configuration which might be used on a Mars orbiter) ranges from 10^{-6} to 10^{-3} g. If one uses Whipple's "best-estimate" curve for meteoroid flux (Ref. 6) and again assumes typical tankage configurations that might be used with a pressurized propellant feed system, the probability of at least one puncture on a year-long mission becomes 0.2, which is intolerably large. Conversely, if one asks how thick the tank walls must be to provide a probability of puncture of less than 1 in 100 for this mission, the Whipple estimates yield a tankage mass about twice that of tanks designed without concern for the meteoroid hazard. This protection can, of course, be provided with less mass by appropriately designed meteoroid shields and by a protective arrangement of other spacecraft parts. Obviously, unless present calculations are revised drastically downward by resolution of the order-of-magnitude uncertainties now existing in the flux and penetration potentials, the meteoroid environment will become a major factor in the design and optimization of propellant tanks.

F. *Propulsion-System Design Choices*

These projected spacecraft-mission requirements and, more generally, the space environmental conditions have led to three basic propulsion-system design choices, which have been used to guide and restrict the scope of the advanced technology being pursued at JPL in the field of Earth-storable propellants. Under the first of the choices, studies of propellant pumping by means of gas-pressurized tankage are being pursued to the exclusion of pump-fed-system investigations. The mass advantages of pump-fed systems are not evident, even for spacecraft systems as large as the one discussed here for the Mars orbiter. For much larger systems, mass advantages are possible by virtue of the lower storage-tank pressures and, hence, the thinner walls of the tanks. However, the uncertainties of the meteoroid environment make thicker tank walls a desirable design feature; in fact, the spacecraft mass optimizations relating tankage mass to meteoroid-shield mass may prescribe tank walls even thicker than would be derived for gas-pressure-fed systems with the tanks designed to sustain only the pressure loads. Aside from these considerations, the great emphasis we are placing on system simplicity would bias us heavily toward the pressure-fed-system options.

Under the second propulsion-system choice, most of the attention in the zero-g starting problem is being directed toward development of flexible, impermeable barriers to separate the liquid and vapor phases in the main propellant tanks. Some work on a new start-tank concept has been initiated, but essentially no effort is being made to develop capillary-action (surface-tension) techniques. In addition to supplying liquid to the engine at start-up, the flexible barrier is believed to be the most positive means for controlling the location of the propellant center of gravity at the critical moment of start-up. Following the first few propulsive maneuvers, the propellant tanks will be only partially filled. If the residual-propellant position is not adequately controlled during zero-g coast, the shift of the spacecraft center of gravity away from the thrust line could result in loss of attitude stabilization, just after engine start but before the propellants become settled by acceleration effects.

Under the third basic system design choice, refractory and ablative thrust chambers are being tested in the hope that regenerative chamber cooling can be avoided. For nonregenerative engines, design simplicity is obtained by reducing the number of seals to a minimum and eliminating a possible need for extra bleed valves or fill valves. Spacecraft operational simplicity is obtained by eliminating a possible requirement for either sequenced starts or prefiring filling and postfiring venting of propellants in the chamber coolant passages (a requirement related to thermal control). In many spacecraft designs, the thrust chamber will be nearly isolated, both thermally and physically, from the main body of mass (see, for example, Fig. 1). This is done to prevent the overheating of sensitive spacecraft elements during motor burn, either by conduction from the hot engine or by radiation or impingement-heating from the highly expanded exhaust plume. Since the chamber is thus thermally isolated, close control of the chamber temperature during coast phases is difficult, particularly during attitude-change maneuvers when shadow patterns are changing. To avoid freezing or boiling of propellants in the chamber passages of a regeneratively cooled engine, either the passages must be vented between firings, a procedure which complicates propulsion-system design and operation, or active thermal-control elements must be employed, thus posing other system complications.

G. *Current Work*

Considerations such as those presented here have led to our conclusion that the Earth-storable propellants have a singular place in the future applications of spacecraft propulsion. The importance of two key aspects of storable-propellant technology is stressed in our current liquid-propulsion research and advanced development programs. First, the desire for complete phenomenological understanding of all elements of a spacecraft propulsion system is the basis for continuing research into the dependence of combustion and chamber-material endurance on propellant injection; a summary of work in this area is presented in Section II. Second, heavy stress is placed on developing simple and highly reliable spacecraft propulsion systems and components; in Section III, two concepts for storable-propellant propulsion systems are reviewed, and some successful developments of advanced component designs are described.

II. COMBUSTION, INJECTION, AND MATERIALS COMPATIBILITY IN LIQUID-PROPELLANT ROCKET ENGINES

Jack H. Rupe
David D. Evans

A. *Introduction*

One of the most pressing problems facing the aerospace industry today is the straightforward development of the complex, sophisticated propulsion systems required for spaceflight missions. Recent history has shown that easily controlled, reproducible systems, characterized by the high performance and extreme reliability which are invariably presumed in preliminary systems design, are rarely, if ever, achieved within our current technology. If one searches for the reasons for these shortcomings, it becomes clear that they can, for the most part, be attributed to a lack of understanding of the truly significant phenomena that dominate such systems and to lamentable deficiencies in the engineering information required as the basis for logical system design. Although these observations are more or less true with regard to all the essential components of a typical system, they are particularly applicable to the problem of injector design and to the achievement of efficient, stable combustion processes that are compatible with a combustor wall.

For a number of years, we at the Jet Propulsion Laboratory have been concerned with these problems. Particular emphasis has been placed on developing the understanding essential to the a priori design of simple, efficient, reliable systems. The degree of our success in meeting these objectives has not been overwhelming, but some progress has been made in providing design principles based on phenomenological understanding. A brief summary of this effort as it applies to combustion, injection, and materials compatibility is presented here, together with a short review of the results obtained by applying the available data to one set of design objectives. Conceptually, at least, these objectives are similar to those established by the requirements of a space propulsion system utilizing Earth-storable propellants in accordance with the premises reviewed in Section I.

Stripped to its bare essentials, the usual problem facing the rocket injector and/or engine designer consists simply of specifying a piece of hardware that will, in a very efficient manner, convert the chemical energy stored in a liquid propellant (or propellants) into the thermal energy of the reaction products. It is also usual to require that the conversion process be achieved in a minimum volume, that it be easily controlled, and that it be extremely reliable and reproducible. It is therefore obvious that the accomplishment of these objectives is dependent on the designer's having at his disposal sufficient quantitative information to allow him to control, through injector design details, those parameters that influence the combustion phenomena.

Thus, if it is determined that an efficient reaction is dependent on the mixture ratio at which the reaction proceeds, then the designer must be able to predetermine and control the *local* mixture ratio. Obviously, control of the overall or gross mixture ratio is insufficient in itself, since the reaction requires mixing to a relatively small scale—ultimately, to the molecular level. Similarly, if it is determined that the efficiency of the reaction is dependent on the mean particle size of the injected propellant, the designer must be able to predetermine the required particle size and then to specify the particular injector configuration that will produce that particle size. This procedure must be continued until all the significant parameters are considered.

The one singular fact that evolves from this assessment of the problem is that the only parameters available to the designer are of a geometrical nature and are contained within the physical configuration of the injector and chamber. Thus, although the ultimate objective is to achieve an efficient, stable combustion process that is compatible with its boundaries, the relative success of an engine design is largely dependent on the designer's ability to correlate, first, the geometric parameters at his disposal with those parameters significant to the prereaction situation and, subsequently, this prereaction situation with combustion phenomena. Note that the parametric description of an injector element, be it of the doublet, triplet, showerhead, or other type, is *not* crucial to good design; any type that will produce the necessary prereaction condition will be successful.

B. Significant Properties of the Prereaction Zone

A complete and general description of the combustion-chamber environment is probably impossible; but, if one concentrates on those prereaction properties that are directly relatable to the injection scheme, it is possible to limit the so-called significant parameters to the mass-flux distribution, the local composition as exemplified by the local mixture ratio, and the local droplet-size distributions. Further, if the characterization of the chamber environment can be restricted to what may be termed the prereaction volume (i.e., that volume in which conditions of the fluid are dominated by the geometry and hydrodynamics of the injection scheme), then the significance of these several terms can be summarized as follows.

1. Mass Distribution

The term *mass distribution* refers to the axial-mass-flow-rate distribution of the one-dimensional flow within a cylindrical chamber. This parameter serves to define the relative propellant concentrations and, on an absolute scale, should presumably define the maximum tolerable concentrations for any given propellant combination. Control and specification of this parameter provide a basis for achieving uniformity in local concentrations and, hence, in axial mass flux in a chamber. Also, when the injector is a composite of a number of essentially identical elements, the properties of the element can be used to construct the gross injection characteristics. This procedure can be utilized to obtain a prescribed injection pattern (mass distribution) since, in most cases, it is simpler to "organize" the mass distribution of a number of small elements to conform to a particular chamber geometry than it is to fabricate suitable chamber boundaries to suit the mass distribution of a small number of elements (i.e., one or two).

2. Mixture-Ratio Distribution

This parameter is simply a measure of the degree of mixing uniformity achieved by the injection processes. Presumably, the ideal situation from a combustion-chemistry viewpoint is attained when a predetermined mixture ratio (such as the peak-performance mixture ratio) is achieved on a molecular scale in a minimum time and/or space. For most engine applications, however, it is probable that a scale of mixing substantially coarser than molecular is sufficiently small. This parameter, in combination with the mass distribution discussed above, can also be utilized to control boundary conditions along the chamber wall.

3. Particle-Size Distribution

This property of the propellant spray is important whenever heat transfer to the propellant surface is the dominant mechanism in elevating the propellant to an exothermic (self-sustaining) reaction temperature. Monopropellant systems are certainly contained in this category, whereas highly hypergolic[2] systems, with reasonable mixing in the liquid phase, can achieve the required reaction temperature without relying on heat transfer from the reaction products and hence may be unaffected by changes in droplet-size distributions.

C. Characterizing the Prereaction Volume

If one accepts these parameters (mass, mixture-ratio, and particle-size distributions) as being the dominant criteria for describing the prereaction environment, it then remains to ensure that their quantitative values will be achieved in an actual combustion chamber. It is obvious that merely evaluating these properties in a completed assembly cannot delineate the requirements for a successful design. Thus, if one is ever to achieve a logical design procedure, it is essential that means be devised for predetermining the expected values for individual parameters and the interactions which lead to desired effects. To this end, a substantial effort at this Laboratory has been based on the premise that the dominant properties of the prereaction environment in a combustion

[2]Propellants that ignite spontaneously as a result of simple mixing at ambient conditions.

chamber can be related to parallel properties of sprays produced with nonreactive liquids. Thus, readily attainable nonreactive-spray information can be utilized as a logical basis for injector design.

Although it is reasonable to expect that combustion will affect all spray properties to some degree, the fact remains that such combustion effects on the properties of the prereaction environment are determinable. Injection into a combustion environment does not alter the possibility that the overall characteristics of the prereaction region may be deduced from the properties of the injected sprays, provided that a quantitative evaluation of the more complicated system can be achieved. Further, the characteristics of the injection scheme can be determined directly, at least for those systems utilizing either relatively nonhypergolic propellants or single-component sprays, where combustion effects can be considered negligible insofar as the initial propellant distributions are concerned.

In an effort to provide a portion of the information required for this design concept, the properties of sprays produced by a pair of unlike impinging jets of nonreactive liquids were studied experimentally (Refs. 7 to 9). A single pair of unlike impinging streams was chosen as the configuration for these studies because: (1) it is widely used as an elemental injector component; (2) it has the potential of substantially enhancing the mixing of unlike fluids; and (3) it is the simplest of a large family of impinging-stream elements, including the coplanar-symmetrical-triplet, the symmetrical-quadruplet, etc. It should be understood, however, that all injection schemes producing equivalent *prereaction environments* must be equally suitable from the standpoint of the combustion process *as long as that environment is controlled, reproducible, and predictable.*

The significance of these latter conditions cannot be overemphasized. Failure to control the injection hydraulics as the all-important boundary condition, whether in experiment, development, or flight applications, ensures failure to develop quantitative correlations. Throughout the ensuing discussion, it is to be remembered that any reference to properties of sprays or the prereaction environment implies an injection scheme that incorporates jets and/or sheets of liquid having steady, reproducible, and predictable (or at least determinable) hydrodynamic properties. In the experiments discussed in Refs. 7 to 9, these requirements were met by using cylindrical jets characterized hydrodynamically by fully developed turbulent flow at the orifice exits, as recommended in Ref. 10.

The experimental determination of the mass and mixture-ratio distribution for sprays of nonreactive liquids was accomplished at this Laboratory by the sampling technique depicted in Fig. 2, which presents a collection of photographs and an artist's conception of several of the more basic injector elements. Despite certain obvious differences, evident in the sketches, the prime objective in every case is to achieve some degree of controlled mixing with a particular mass distribution; further, in every case, the element depends on the hydrodynamic properties of free-liquid sheets or jets to accomplish this objective. Thus, the control of the hydraulic properties of these elements is prerequisite to the control of mass and mixture-ratio distributions. It is pertinent to reiterate here that, once the required properties of an injection scheme have been defined, any or all such elements could be utilized to achieve those requirements. It is only because the properties of sprays formed with *unlike impinging streams* have been evaluated in some detail that this element was utilized in these investigations.

Figure 2 (b) shows two views of a typical spray produced by impingement of a pair of nearly identical water jets. Note that the bulk of the spray is concentrated about a *resultant momentum line* and has, in this case of identical jets with equal momenta, a nearly elliptical cross section. Now, if a collector of the type shown in Fig. 2 (c) is exposed to such a spray for a reasonable time interval, a series of samples such as those shown in Fig. 2 (d) can be obtained. In this case, the vertical height of the sample in each tube is proportional to the mass flux at a different position within the spray, thus permitting direct evaluation of the mass distribution produced by the spray. In addition, if the injected fluids are immiscible, they will separate after the sample is obtained (as indicated in the photographs) and will provide a measure of the relative flow rates at a particular point in the spray. These data, together with additional samplings, will yield the mixture-ratio distribution.

A large amount of this kind of information, obtained with the carbon tetrachloride–water system, was utilized to produce a correlation between a quantity η_m, known as the *mixing factor*, and the gross dynamic properties of

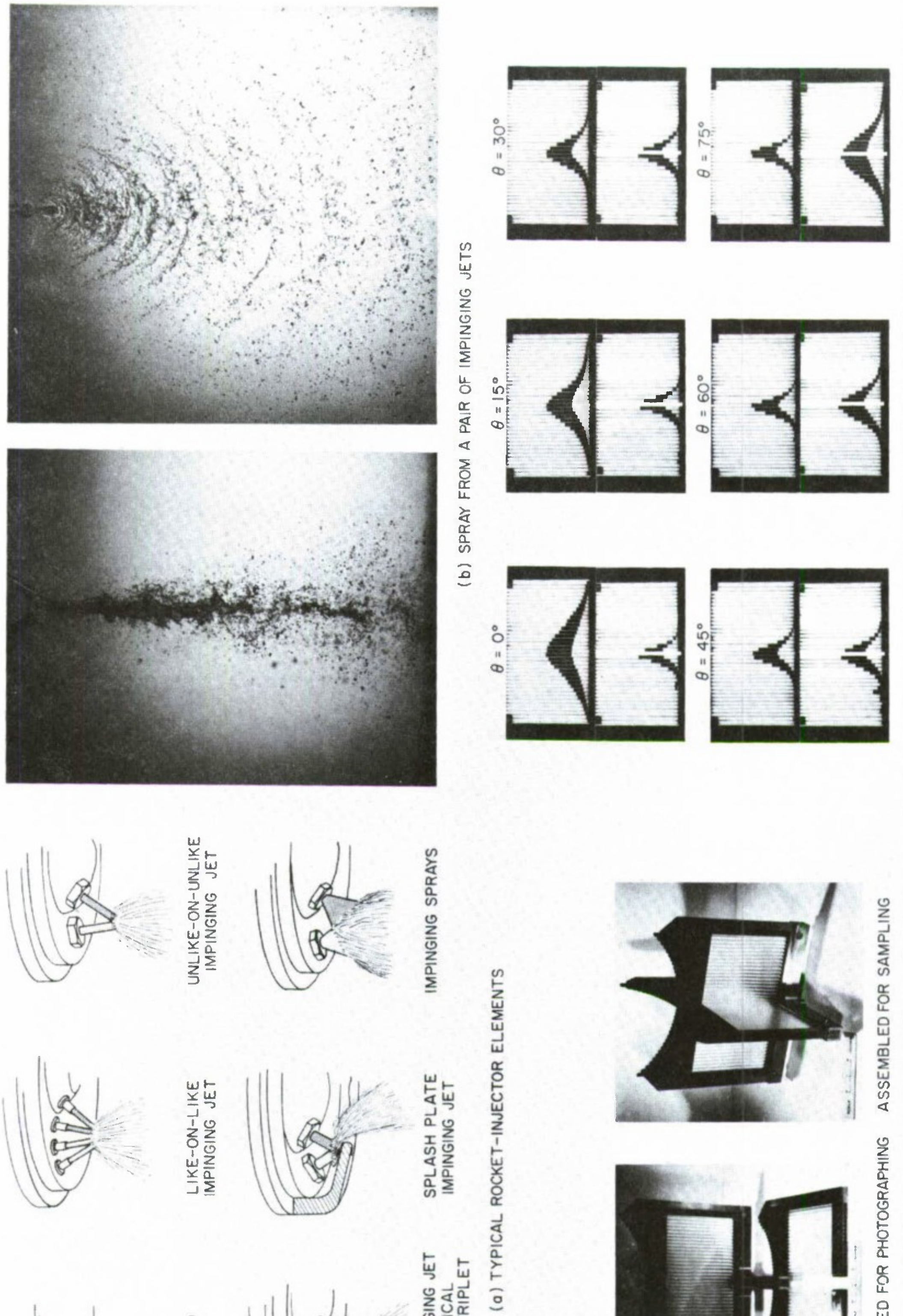


Fig. 2. Montage of nonreactive-spray experiments

the two jets (Refs. 7 and 8). The mixing factor, which has limiting values of 0 and 100, is a measure of the mixing efficiency of the spray. In at least one sense, this quantity can be imagined to represent the percentage of the total spray that has achieved the intended mixture ratio.

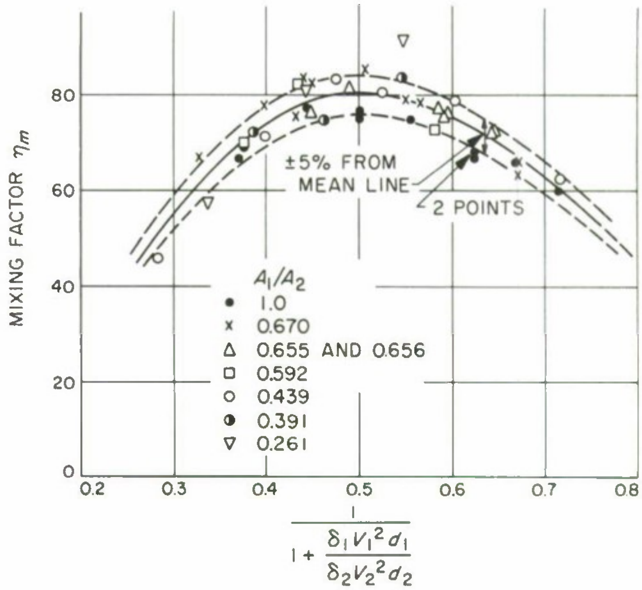
A correlation resulting from this effort (Fig. 3 and Ref. 8) formed the basis for the conclusion that (within the limitations of the experiments) the most uniform mixture-ratio distribution is achieved in the spray produced by a pair of impinging streams when the products of fluid density δ, velocity squared V^2, and jet diameter d are equal for both streams, or when

$$\frac{\delta_1 V_1^2 d_1}{\delta_2 V_2^2 d_2} = 1.0$$

This quantity, η_m, has become known as the *uniformity criterion*, although it might more appropriately be termed the *mixture-ratio* or *mixing-uniformity criterion* to distinguish it from mass-flux uniformity. It should be noted that the correlation in Fig. 3 is based on independent variations of velocity, diameter, and density, as well as the ratios of these quantities.

If, in addition to the usual mixture-ratio specifications, it is required that the element satisfy this mixing-uniformity criterion, then the orifice diameter ratio and the jet velocity ratio for any given propellant system are defined, respectively, by Eqs. (c) and (d) of Fig. 3. If it is further assumed that the total flow rate for the element $\dot{w}_{el}$ is to be determined from other considerations, then Eq. (e) of Fig. 3 must also be satisfied. Obviously, then, the additional arbitrary choice of one velocity or one diameter will determine all other values. In any event, when Eqs. (c) and (d) are satisfied, the element which incorporates these two specified ratios will produce a spray having a near-uniform mixture-ratio distribution, so that at least this aspect of the design is delineated.

It has been shown that similar correlations are possible for the coplanar-triplet element and certain other element geometries (Ref. 11); experimentally, however, the mixing-uniformity criterion tends toward the same relative level: i.e., maximum η_m = 85 to 90. Thus, any advantages to be derived from the use of these relatively complex elements must stem from some other consideration.



For the maximum value of η_m (i.e., near-uniform r distribution), the uniformity criterion is expressed by

$$\frac{\delta_1 V_1^2 d_1}{\delta_2 V_2^2 d_2} = 1.0 \qquad \text{(a)}$$

and, by definition,

$$r = \frac{\delta_2 V_2}{\delta_1 V_1}\left(\frac{d_2}{d_1}\right)^2 \qquad \text{(b)}$$

Therefore, combining Eqs. (a) and (b) gives

$$\frac{d_1}{d_2} = \left(\frac{\delta_2}{\delta_1}\right)^{1/3}\left(\frac{1}{r}\right)^{2/3} \qquad \text{(c)}$$

and

$$\frac{V_1}{V_2} = \left(\frac{\delta_2}{\delta_1} r\right)^{1/3} \qquad \text{(d)}$$

For a particular flow rate,

$$V_1 = \frac{576\, \dot{w}_{el}}{\pi \delta_1 d_1^2 (1+r)} \qquad \text{(e)}$$

Note also that, for design conditions where r, δ_1, δ_2, and d_1/d_2 satisfy the uniformity criterion,

$$\frac{P_1}{P_2} = \frac{d_1}{d_2}$$

where P is momentum per second. Thus, for any given injector, the value of r which satisfies the uniformity criterion is expressed by

$$r_{unif} = \left[\frac{\delta_2}{\delta_1}\left(\frac{d_2}{d_1}\right)^3\right]^{1/2}$$

Fig. 3. Mixing-uniformity criterion for unlike impinging doublets

Unfortunately, no simple way of characterizing the mass distribution of the spray produced by an element has been devised. This problem is particularly difficult, since such distributions tend to be strong functions of the relative geometry and dynamic properties of the jets (even those which satisfy the mixing-uniformity criterion), as well as the included angle between the jet centerlines (Ref. 7). In designing a composite injector, the usual procedure is to specify the impingement angle on a nearly arbitrary basis, since it has a relatively small effect on mixing efficiency. It is then possible to utilize experimental information, obtained with an actual injector element similar to the proposed design, to acquire the desired mass-flux distribution. This procedure is permissible, since the geometrical properties of the sprays produced by a pair of jets having similar impingement angles, orifice diameter ratios, and jet velocity ratios tend to be relatively insensitive to scale and absolute levels of mass flow rates (Ref. 7). It is possible, therefore, to approximate the mass distributions of a proposed element from other information that may be available: e.g., from the experimental data used to determine the mixing correlation.

One of the techniques developed for the purpose of characterizing the mass-flux distribution of an element is illustrated in Fig. 4. In this instance, the original measured values of mass flux vs position along a great circle of the sampling surface are plotted to give smoothed curves through the experimental data. These curves are used to locate lines of constant mass flux on the spherical surface at which data have been obtained. The contours are then projected along radii emanating from the impingement point to a plane perpendicular to the resultant momentum line. The values associated with these contours are subsequently corrected for variations introduced by the projection and to give the axial component. These modifications yield the constant-mass-flux contours, shown as light lines in the Figure. The spray boundary shown here is located in the projection plane and indicates the iso-mass line that encloses 95% of the total mass flow of the spray.

Superimposed on the mass-flux distribution in Fig. 4 are lines of constant local mass-fraction ratio (i.e., mixture ratio) as determined at the same location within the spray. It is seen that, characteristically, the contours are parallel lines which are generally perpendicular to the plane of the stream centerlines, illustrating the so-called penetration phenomenon in the plane of the jet centerlines: i.e., fuel-rich relative to the fuel source in the area of spray *across* the resultant momentum line from the fuel source.

Although the mass-flux distributions produced by these elements are adequately represented by the contours of Fig. 4, a "three-dimensional model" of such distributions has also been devised, wherein the photographic-transmission density of a negative is proportional to the axial mass flux. It was intended that this model would provide a quantitative description of the mass distribution achieved in a combustion chamber, and that, within limits, it would be useful in representing local mass fluxes from the combined flows of several elements. A photographic reproduction of the mass-flux analogs (as a positive rather than a negative) for the same element is shown in Fig. 5, and a typical application of this concept to a design for a small 10-element injection scheme is illustrated in Fig. 6. The techniques and procedures utilized in generating these models are presented in detail in Ref. 12.

In referring to Fig. 6, it should be noted that the combined distribution is illustrated at a station which has been designated as the *model plane*. The combined distribution comprises several of the mass distributions for a single element, as shown in Fig. 5, but the "scale" has been reduced to conform to the following definition: at the model plane, each element must contribute axial mass flow to an area equal to $1/N$ times the chamber cross-sectional area, where N is the number of identical elements incorporated in the injection scheme. Also, note that the absolute values of axial mass flux are determined by the flow rate required from each element as scaled to the total engine flow rate. Thus, at the model plane, if each element were characterized by a uniform axial mass flux, and if the boundaries of the spray formed by each element were such that they could fit together like pieces of a jigsaw puzzle bounded by the chamber circumference, then the axial-mass-flux distribution at that station would be uniform.

A change in the mass-distribution scale for a given element is permissible, since it is reasonable to assume that the spray particles travel along radial lines which emanate from a source very near the impingement point. Thus, if aerodynamic and body forces are small, the relative distribution at various distances from the impingement point can be predicted from simple geometrical relationships.

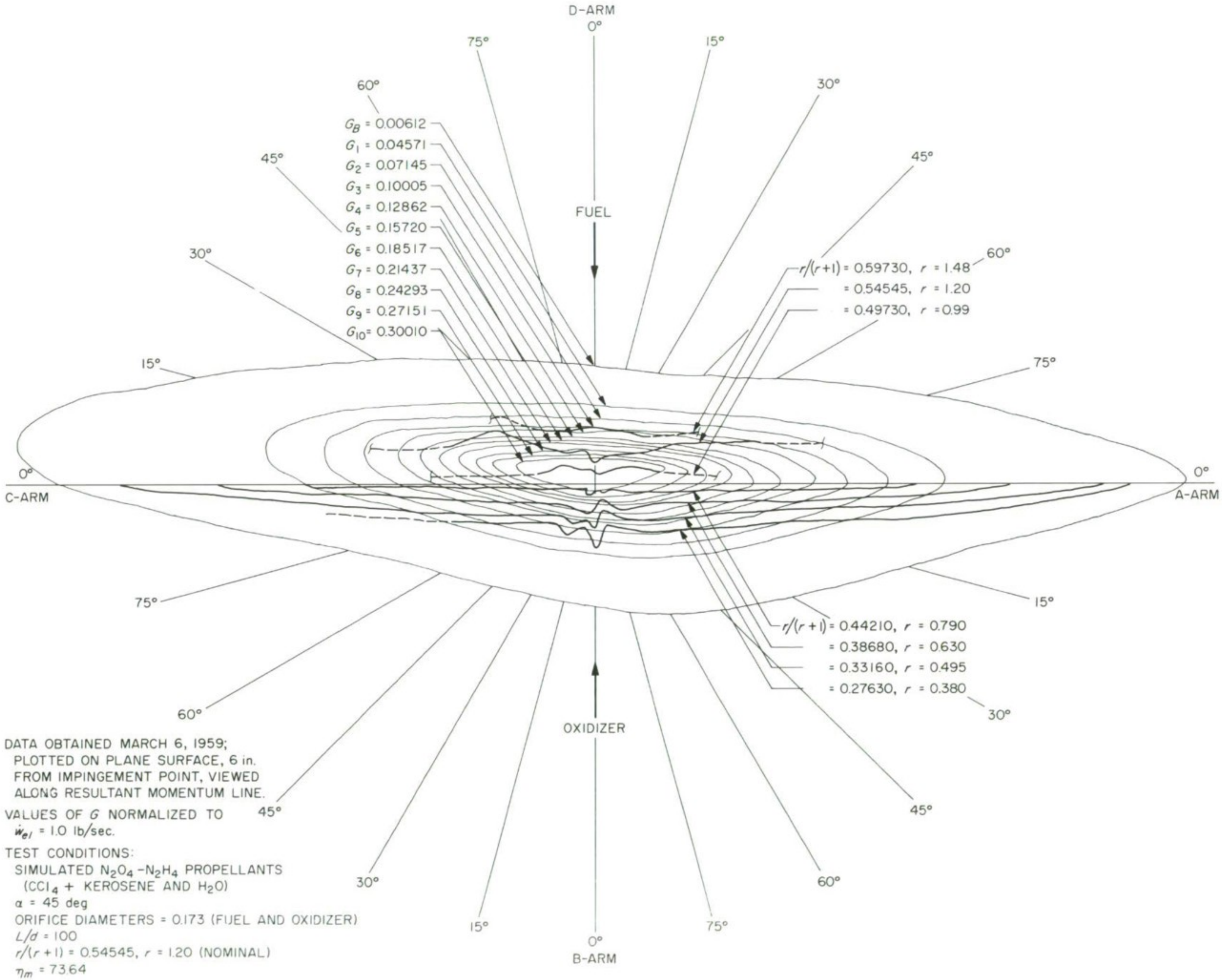


Fig. 4. Mixture-ratio distribution for one configuration of an unlike-impinging-doublet element

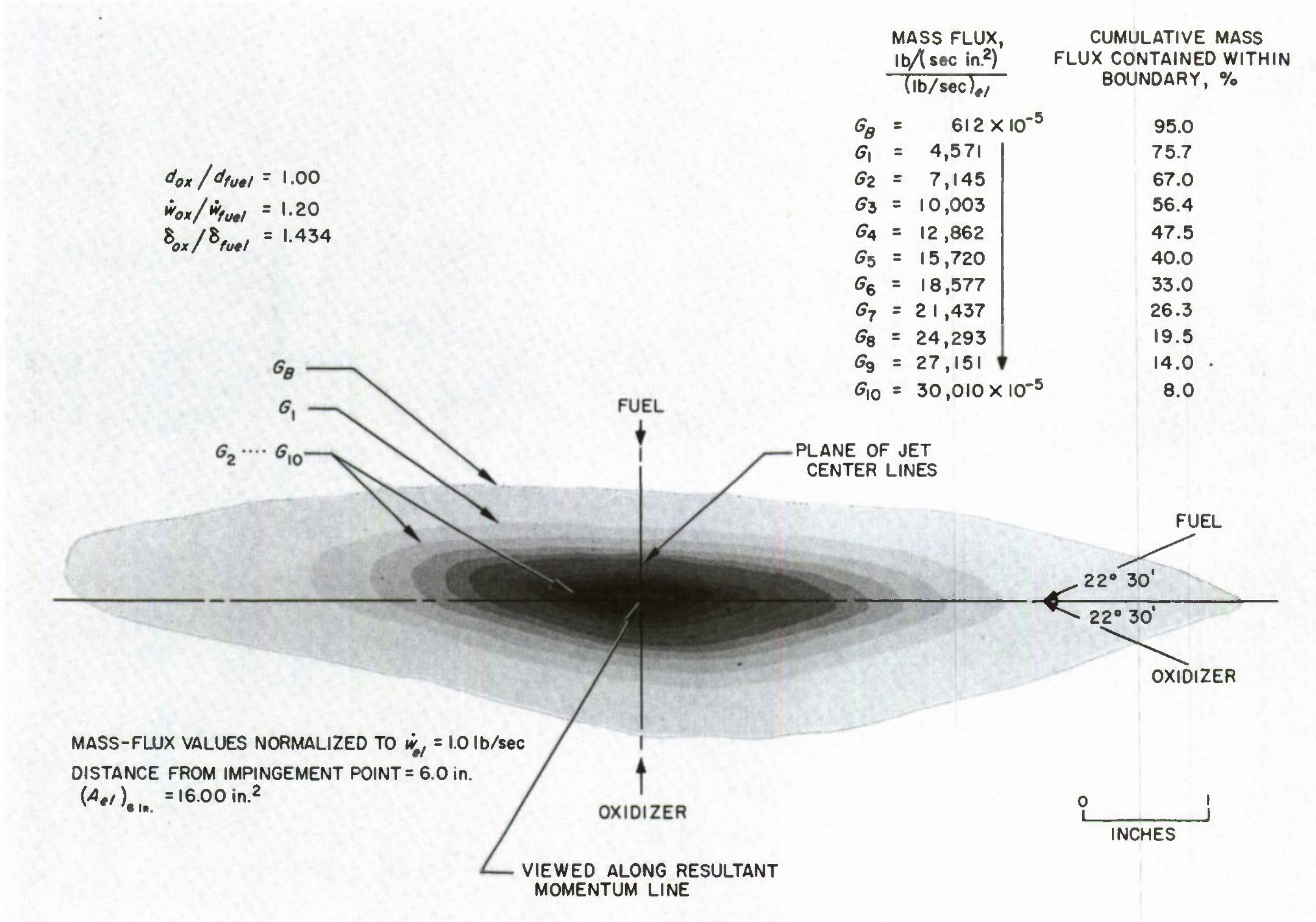


Fig. 5. Analog of mass-flux distribution for one configuration of an unlike-impinging-doublet element

D. A Composite Injector Design

The mass-flux distribution illustrated in Fig. 5 was prescribed for an injection scheme to be used in the evaluation of combustion-chamber materials. It was intended to present to the chamber wall a variation in boundary-flow composition, as well as in mass flux. As is evident in Fig. 5, a substantial interaction with the wall very near the injector can be expected in the regions near "5 and 11 o'clock", even to the point where direct liquid impingement is a possibility. Further, assuming that the propellant composition predicted by studies with nonreactive fluids is unaffected by combustion, one would expect a *fuel-rich* zone near 8 o'clock and an *oxidizer-rich* zone near 2 o'clock. In any one test, the chamber wall would thus be exposed to a variety of environments which should encompass nearly all the pertinent variations. Hence, a true evaluation of the suitability of a wall material is possible. Further, if *variations* in the lifetimes of the materials are observed, it should be possible to specify a boundary environment for which compatibility is demonstrated and also to prescribe the injection characteristics that could produce a compatible environment.

Gross characteristics determined in this way were verified when an actual injector, identified as the *10-element Mod II*, was constructed and used to obtain the nonreactive-spray properties of the combined flow and its compatibility with a chamber wall during actual firing conditions. This injector was designed to produce 100 lb of thrust when operating with the nitrogen tetroxide–hydrazine propellant combination at a mixture ratio of 1.2 and a combustion-chamber pressure of 150 psia. As indicated in Ref. 13, where these results are presented in detail, the areas of severe erosion were found to correspond with those of high local heat flux

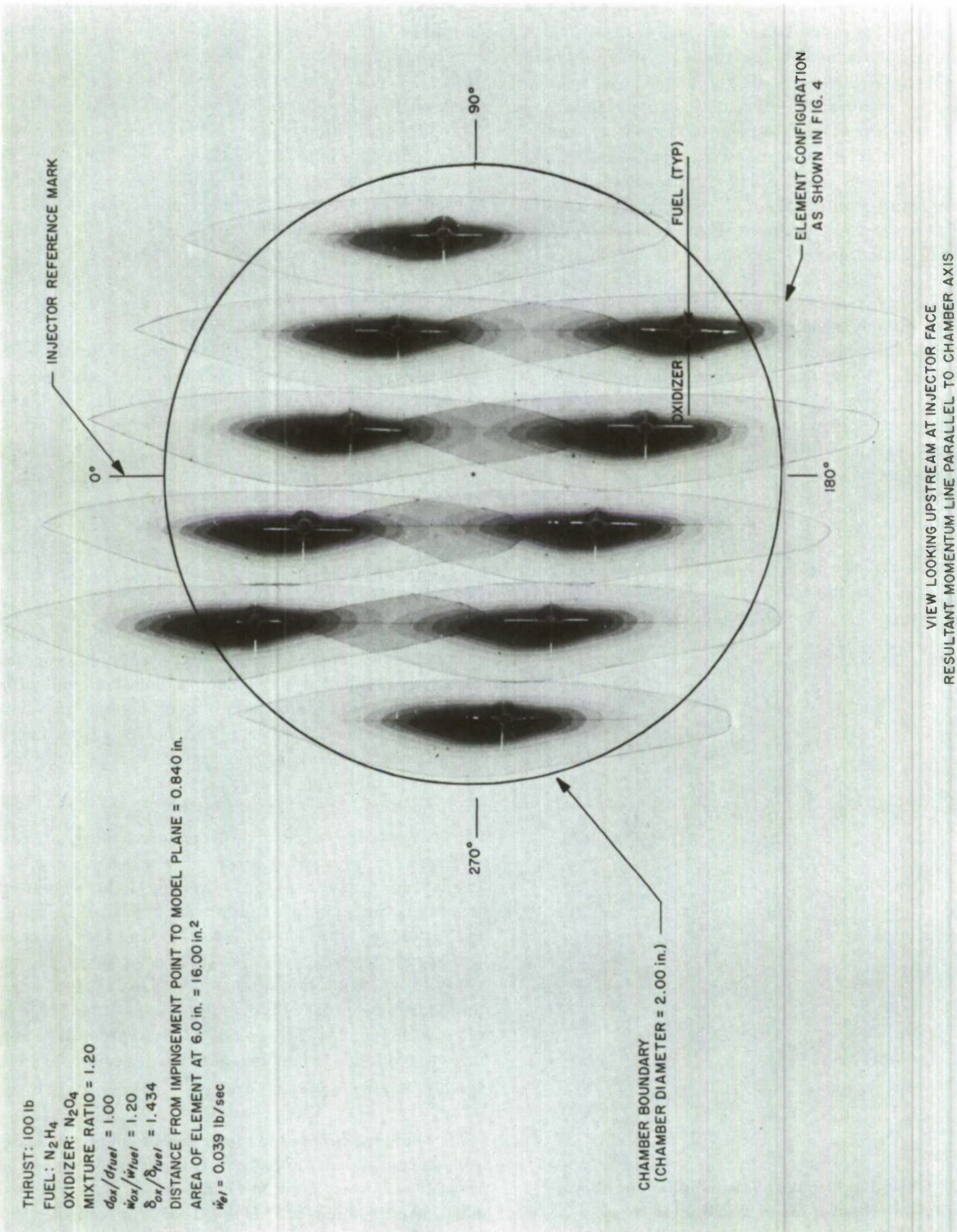


Fig. 6. Mass-flux distribution for the 10-element Mod II injector

for materials like the Refrasil-phenolics, or to regions of high local mixture ratio at the wall (i.e., oxidizer-rich as determined from the nonreactive studies) for composition-sensitive materials like pyrolytic graphite. The correlation between local heat-transfer rates and ablation is indicated in Fig. 7, which presents a typical eroded-throat configuration on which is superimposed a heat-transfer distribution. Obviously, high ablation rates correspond to high heat-transfer rates; these, in turn, can be related to local mass and mixture-ratio distributions which are controlled by the injection scheme.

Although it is clear from Fig. 7 that particular regions exhibited relatively severe erosion rates, one of the more significant results of this effort is the observation that there was also a substantial region where the erosion rate was very low. It must be concluded that, if the conditions prevailing in the low-erosion region could be reproduced over the entire nozzle surface, then relatively long nozzle lifetimes could be achieved.

The information gained from these tests was utilized as the basis for a redesign of the injector in order to provide a boundary flow similar to that which had exhibited compatibility with the wall. This was accomplished by reorienting the impinging-jet elements so that

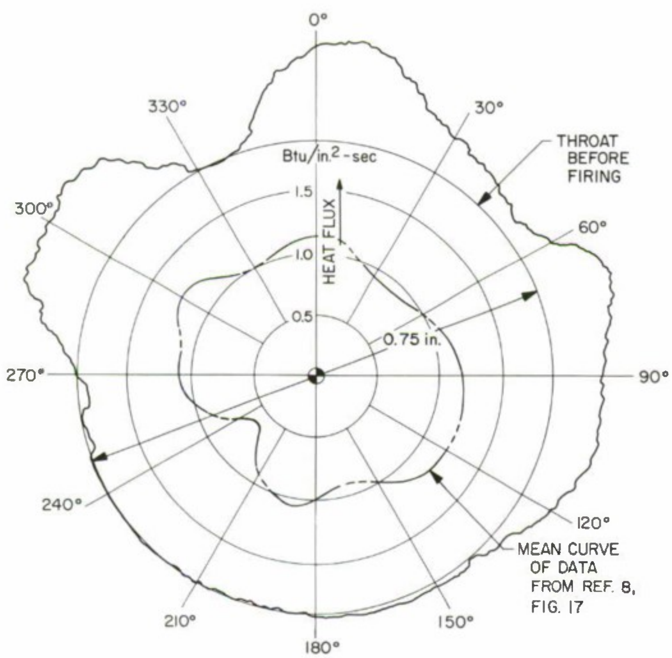


Fig. 7. Erosion in throat of Refrasil-phenolic nozzle related to heat-transfer distribution in test with 10-element Mod IV injector

their resulting spray fans were located in a more uniform, chord-like manner, as shown in Fig. 8. All operating conditions for this injector, designated as the *10-element Mod IV*, were identical with those described above for the Mod II unit. Data from nonreactive spray tests permitted the prediction that a gross improvement in erosion characteristics would be obtained with this injector. Indeed, the erosion characteristics were considerably improved, as evidenced by the cross-sectioned view of a similar Refrasil-phenolic nozzle after a 30-sec firing (Fig. 9). A comparison of these results with those of Fig. 7 reveals nearly an order-of-magnitude reduction in the erosion rates and serves as clear evidence that a controlled, reproducible boundary flow is essential to a meaningful evaluation of material compatibility. In addition, the data demonstrate that the nonreactive-spray properties of injection schemes can be invaluable in providing the basis for the design of controlled, reproducible injection schemes that must produce both high performance and an environment compatible with a chamber wall.

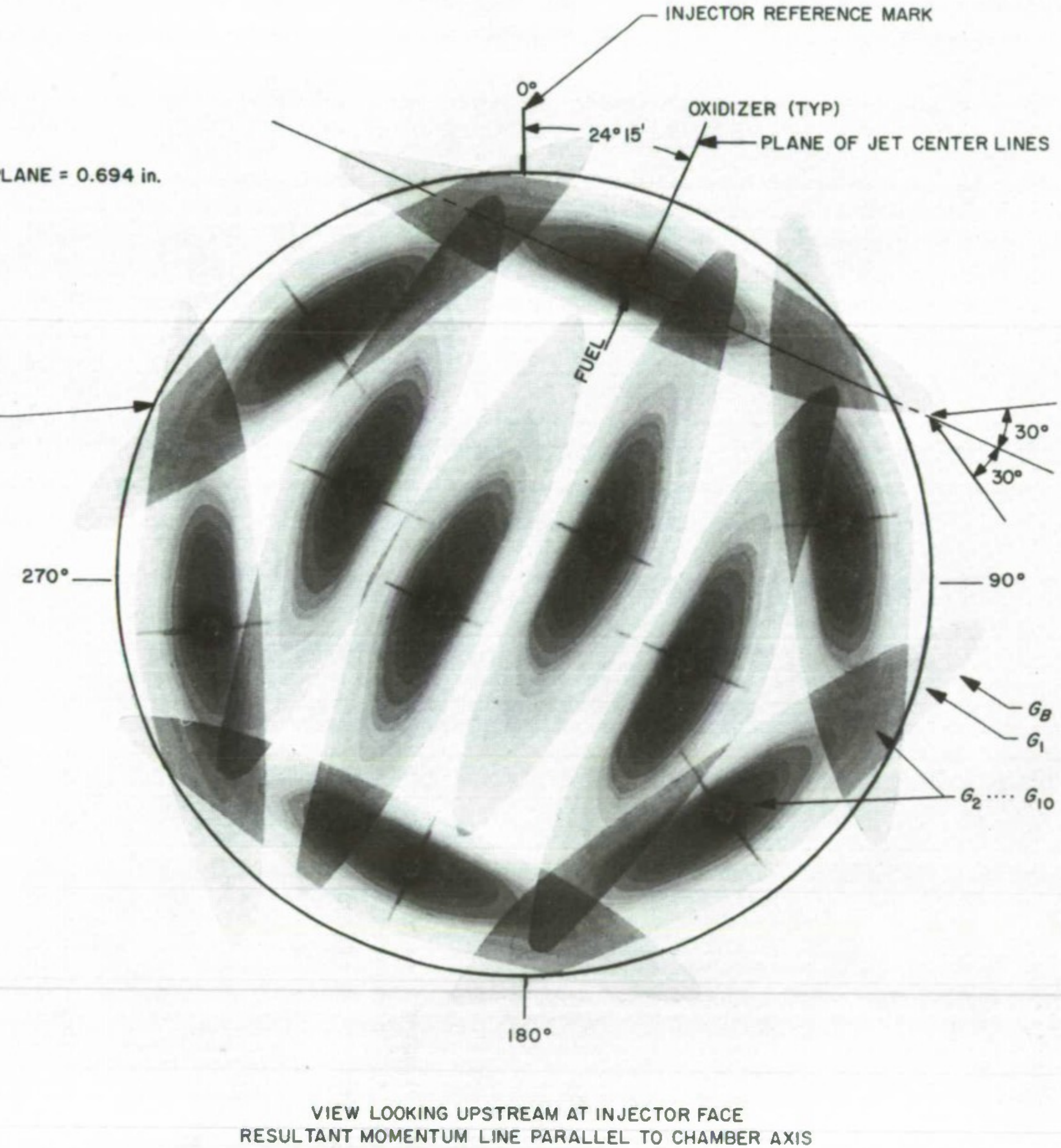
E. Evaluation of Thrust-Chamber Materials

The excellent low-erosion characteristics of this Mod IV injector were also exploited in conjunction with a thrust-chamber-materials development and evaluation program. Tests were conducted with ablatives, both with and without throat inserts, and composite or layered pyrolytic graphite structures; major emphasis, however, was placed on studies of free-standing chambers of pyrolytic graphite (PG) and its alloys.

The pyrolytic graphite thrust-chamber program was of particular interest because of the excellent high-temperature properties and extremely light weight of this unique material. Test durations at the 100-lb thrust level advanced from 30 sec with the Mod II injector to several minutes with the Mod IV injector in an alloyed pyrolytic graphite thrust chamber. During the long-duration tests, throat erosion rates were of the order of tens of microinches per second. This represents a significant reduction when compared with commonly encountered erosion rates.

Since an important additional requirement for many applications is that the exterior-wall temperature of the thrust-chamber assembly must not exceed some nominal value, usually about 500°F, the designs investigated include: (1) a free-standing pyrolytic graphite chamber

THRUST: 100 lb
FUEL: N_2H_4
OXIDIZER: N_2O_4
MIXTURE RATIO = 1.20
d_{ox}/d_{fuel} = 1.00
$\dot{w}_{ox}/\dot{w}_{fuel}$ = 1.20
$\delta_{ox}/\delta_{fuel}$ = 1.434
DISTANCE FROM IMPINGEMENT POINT TO MODEL PLANE = 0.694 in.
AREA OF ELEMENT AT 6.0 in. = 23.6 in.^{2}
$\dot{w}_{el}$ = 0.039 lb/sec



MASS FLUX, $\frac{\text{lb/(sec in.}^2)}{(\text{lb/sec})_{el}}$	CUMULATIVE MASS FLUX CONTAINED WITHIN BOUNDARY, %
G_B = 605 × 10^{-5}	95.0
G_1 = 1,868	82.6
G_2 = 2,839	75.0
G_3 = 3,862	66.4
G_4 = 4,857	60.0
G_5 = 5,605	54.4
G_6 = 6,203	49.6
G_7 = 7,547	40.7
G_8 = 9,414	29.3
G_9 = 10,762	18.0
G_{10} = 11,558 × 10^{-5}	14.0

Fig. 8. Mass-flux distribution for the 10-element Mod IV injector

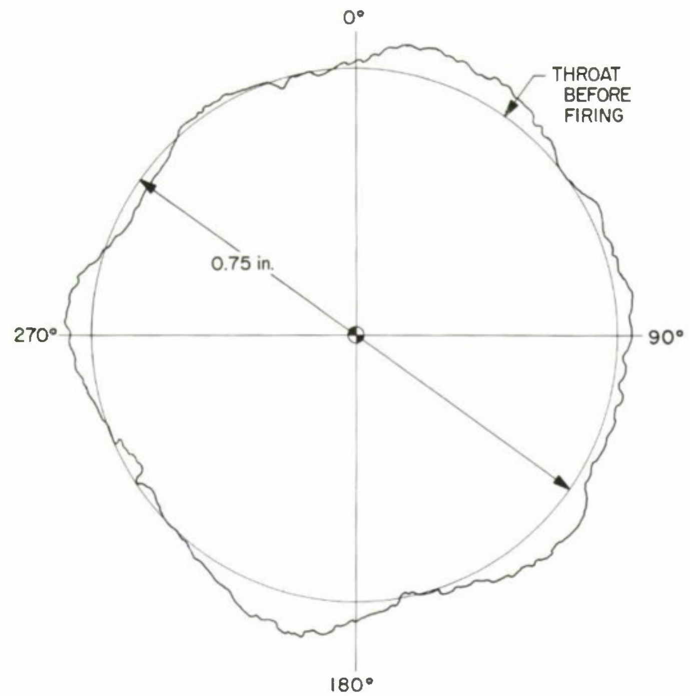


Fig. 9. Nozzle-throat boundary of Refrasil-phenolic ablative thrust chamber after testing with 10-element Mod IV injector

surrounded by a standoff heat shield, and (2) a composite PG structure. In each case, it was demonstrated that the Mod IV injector provided a compatible, controlled boundary flow, as evidenced by the uniform low-erosion rates. Even though erosion rates were small in the composite model, it was demonstrated that this approach was not as effective as the free-standing-chamber concept.

F. Rapid-Reaction Effects on Sprays

Evidence such as that presented above provides verification of the premise that the properties of nonreactive sprays can be used as a general and universal basis for the design of liquid-propellant combustors. However, experience has shown the range of applicability of these concepts is limited to those cases where the effects of the combustion process on the propellant distribution are small. For nonhypergolic propellant systems, there are indications that these techniques may be generally applicable. However, for hypergolic Earth-storables (more specifically, hydrazine-based fuels and oxidizers of the nitrogen oxide type), there is evidence that, under some conditions, the rapid liquid-phase reactions do exert a considerable influence on the mixing process.

The influence that such effects can have on combustor performance was strikingly illustrated during tests with a 2000-lb-thrust single-element injector, incorporating an unlike impinging doublet and designed for nitrogen tetroxide–hydrazine bipropellants at a chamber pressure of 150 psia. Although nonreactive mass and mixture-ratio data indicated that high efficiencies should be obtained, a performance evaluation revealed that the mixing actually achieved by this injector was grossly inefficient, since only about 67% of theoretical combustion efficiency was realized. In an attempt to understand the reasons for this poor performance, an experiment was designed to explore an hypothesis stated by Elverum (Ref. 14), in which the poor mixing of N_2O_4–N_2H_4 propellants was attributed to the degradation of mixing which resulted from the rapid liquid-phase reactions occurring at the interface produced by the impingement of the injected propellant streams.

Briefly, this experiment (described in detail in Ref. 15) involved the firing of a single doublet injector in a cham-

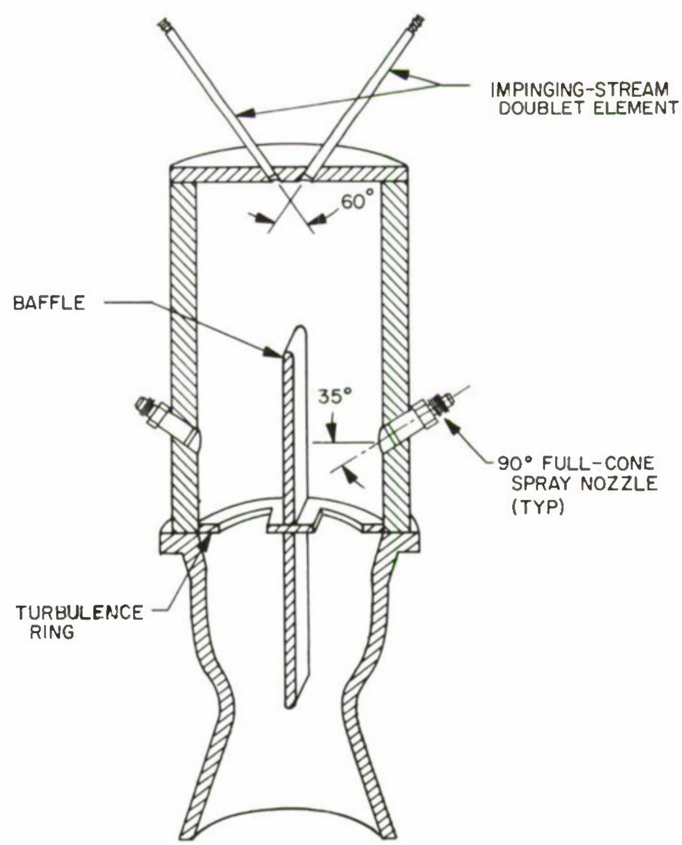


Fig. 10. Experimental apparatus for evaluating combustion effects in sprays

ber which was divided into two longitudinal channels by a baffle plate, as shown in Fig. 10. Two spray nozzles were located in the chamber wall, one on each side of the baffle. The impingement process, together with any effects of combustion on that primary process, determined the initial propellant distribution, and the baffle prevented subsequent secondary mixing due to turbulence and diffusion. Thus, if the composition of the spray in one channel were different from that in the other channel, the spraying of oxidizer into fuel-rich gases and fuel into oxidizer-rich gases ("unlike" propellants) should have increased performance; the spraying of fuel into fuel-rich gases and oxidizer into oxidizer-rich gases ("like" propellants) should have reduced performance. If the streams from the main injector were well mixed, and if a nearly uniform mixture-ratio distribution existed in the chamber, performance should have remained relatively unchanged when the propellant sprays were reversed.

The difference in performance measured for the alternate positions of the propellants in the side sprays was determined at the 2000-lb thrust level with single elements having 0.236-in.-diameter streams, and at the 100-lb thrust level with 0.064-in.-diameter streams. Results of these tests indicate that reaction effects exerted such a pronounced influence on the liquid-phase mixing of the two propellant streams that considerable quantities of fuel and oxidizer did not undergo mixing and combustion, but rather were repelled from each other, presumably by the gas evolvement at the liquid-to-liquid interface. The magnitude of this effect was more pronounced at the 2000-lb thrust level than at the 100-lb level; this is shown in Fig. 11, a graph of combustion efficiency as a function of percentage of side flow. For the 0.236-in.-diameter streams, the addition of 25% unlike-propellant side flow raised the combustion efficiency from a value of 62% to a value of 84%, an increase of 22%. For the 0.064-in. streams, a similar side-flow condition resulted in a 13% increase in combustion efficiency. Thus, the initial evidence indicates a strong "scale" effect, which one can rationalize by noting that large streams have relatively long contact times along the impingement interface, so that gas evolvement on that surface can play an important role in the subsequent propellant distribution.

G. Current Work

Additional experiments with 0.020-in.-diameter streams, planned for the near future, are expected to verify the

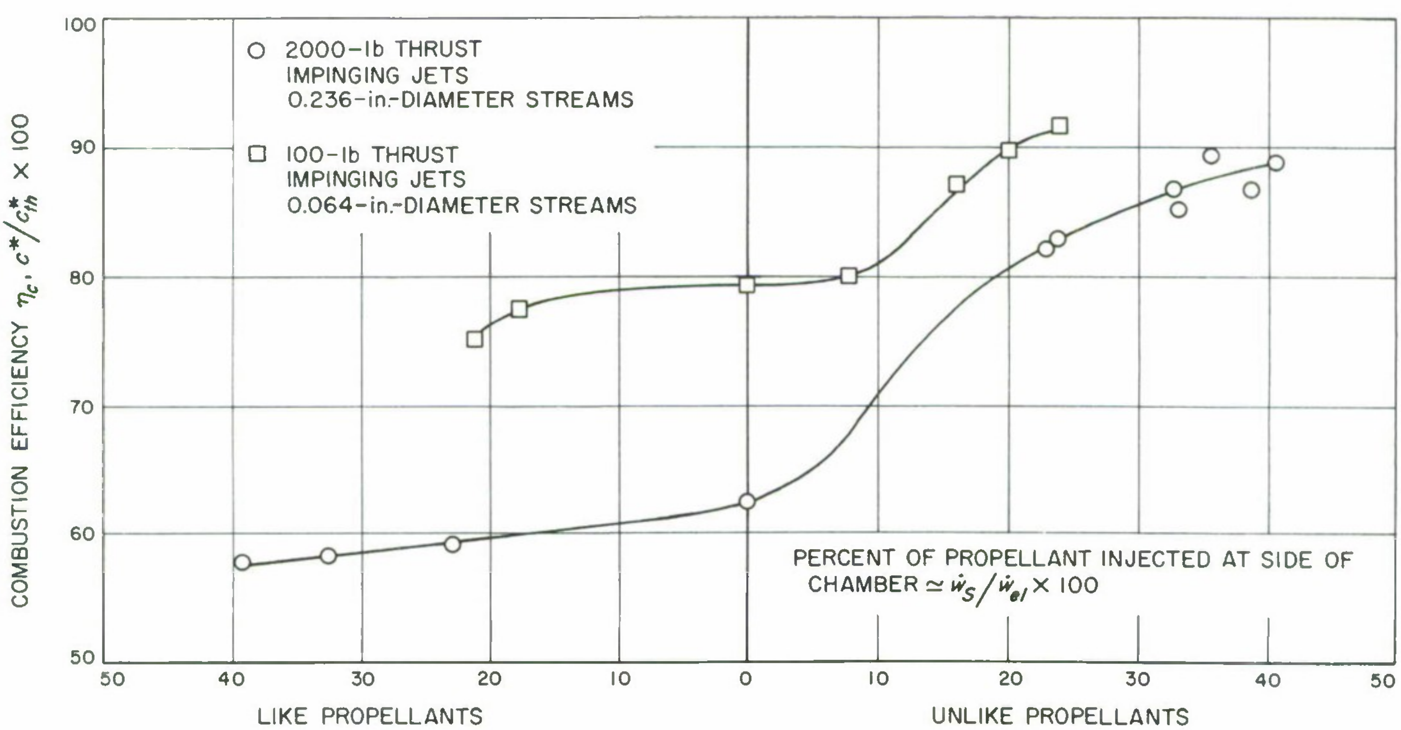


Fig. 11. Results of combustion-effects experiments at two thrust levels

extrapolation of the data of Fig. 11 and also the other evidence indicating that, at 10 lb of thrust per element, the data generated with nonreactive sprays actually do provide an adequate basis for predicting mass and mixture-ratio distributions under combustion conditions.

In addition to the scale effect noted above, it can be expected that the absolute-pressure level will have first-order effects on the degree to which such liquid-phase reactions contribute to propellant distributions. Additional investigations of the effects of that parameter are planned.

III. SOME ADVANCED DEVELOPMENTS IN PROPULSION SYSTEMS FOR UNMANNED SPACECRAFT

Richard N. Porter
David D. Evans

A. Introduction

Considerable effort is currently under way within the rocket industry to develop technology which will satisfy the requirements for spacecraft on-board propulsion. This Section outlines some of the advanced developments in the monopropellant and bipropellant rocket work currently in progress at JPL.

The work described here had its origins in the 1950's, when the characteristics of an "idealized" liquid rocket were under discussion. These talks resulted in the crystallizing of some opinions about criteria by which the merits of advanced concepts could be measured. At the same time, a philosophy of approach to rocket design was generated. Shortly thereafter, a system concept was evolved which satisfied most of these criteria and seemed to provide most of the functional capabilities that spacecraft propulsion systems would need: long-term storability in space, the capacity for multiple starts in zero-g, and provisions for engine throttling. Formal approval to develop this concept was obtained in 1960, and the effort has been continuous since that time.

The program is designated by the title *Advanced Liquid Propulsion Systems* (ALPS). In this program, a conscious effort is being made to follow the design philosophy discussed in Section I of this Report. Major emphasis in this approach is placed on simplicity and control of pertinent physical phenomena. The entire effort is integrated on a system basis so that all the solutions to individual problems will be mutually compatible. Where necessary, special investigations are allowed to develop needed basic data and methods, but the ultimate purpose of implementing the overall system concept is kept paramount.

B. The ALPS System Concepts

The basic ALPS system concept is shown diagrammatically in Fig. 12. It features a variable-thrust rocket engine, fed by two positive-expulsion devices contained within a single vessel pressurized by gas generated through the decomposition of a monopropellant. This system was devised in an attempt to maximize reliability, versatility, controllability, and storability, and to minimize interactions with the rest of the spacecraft.

The principle of achieving reliability through simplicity is firmly applied throughout the system. In addition, an attempt is made to avoid the use of components which have poor reliability records, such as pressure switches, check valves, high-pressure-gas shutoff valves, gas-pressure regulators, and any components operating at cryogenic temperatures. The necessity for electrically operated components is kept to a minimum. Certain "problem" components are incorporated in the system, but only where it seems that sufficient multiple benefits accrue from their use to overbalance their disadvantages. In addition, the prescribed limits of operation (temperature range, pressures, etc.) are as narrow as system simplicity, functional capability, and realistic spacecraft conditions allow.

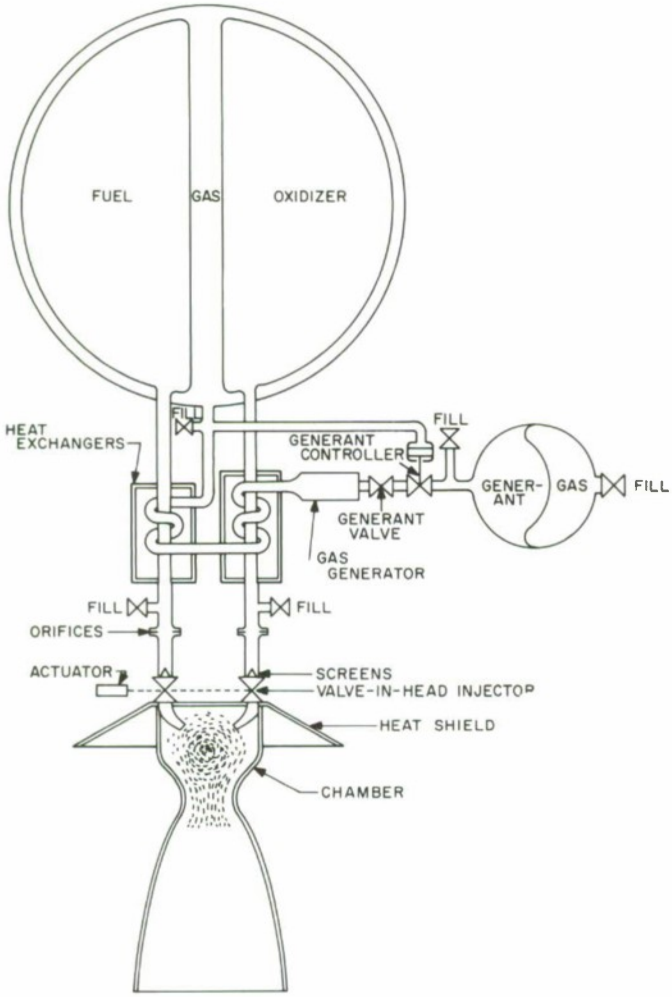


Fig. 12. Schematic diagram showing all components in the ALPS system

Versatility is gained primarily by designing into the system functional capabilities which match the requirements of as many different space missions as possible. For example, the throttling and multiple-start capabilities allow the use of a single propulsion system to provide not only the large impulses required for such retrograde maneuvers as orbiting, rendezvous, and loading, but also the small impulses needed for midcourse trajectory corrections and orbit trimming. The gas generator which pressurizes the propellants can also serve as a source of either hot (1500°F) or cold (150°F) gas, which might be used for such auxiliary applications as attitude control, driving a turbo-alternator, thrust-vector control, inflating extensible structures, separating the spacecraft from a capsule, etc. Potential usage of other parts of the system or the component technology is also an important consideration, since the complete system, per se, may never be used; actual cases in which the concepts or technology have already been used in other programs are cited below.

Control is simplified by eliminating the need for sequencing. On–off control requires only the action of the injector (propellant) valves and a generant valve. Transients are tightly controlled by minimizing the manifold volume downstream of the injector valves. The thrust level is set by an open-loop thrust control, wherein accelerometer signals are used to command the proportional-type injector valves to open farther or to close slightly. The rate at which gas is generated for tank pressurization is controlled simply by the feedback of tank pressure to the pressure-sensing diaphragm in the generant controller.

Storability is enhanced by minimizing the potential for leakage. All active valves seal *only* liquids. The gas is contained by two fill valves and four static seals; nitrogen gas at 1500 psi presents the most severe leakage problem. Storability is further enhanced by using only compatible metals for all parts, with the exception of Teflon in some valves and Teflon or compatible elastomers in the expulsion devices. In contact with such materials, the Earth-storable propellants can be kept in space for years without excessive corrosion or need to vent the tanks.

Interactions with the spacecraft are minimized by making the system as passive as possible during the non-operating periods. This is done by eliminating the need for purges or venting and by the use of propellants which can be stored at the same temperature as that required by the rest of the spacecraft. Also, it is hoped that good control over the center of gravity will be maintained by the mechanical expulsion devices. The system is to be capable of instantaneous activation without reliance on auxiliary systems such as ullage-settling jets.

The preceding design objectives are presumed to be the major conceptual advantages of the system. Realization of these objectives is the goal of the ALPS program. At this writing, no complete ALPS system has been assembled; however, a considerable amount of analytical work has been done, in which analog and digital computer programs have been used to predict the center-of-gravity excursions, the variations in operational parameters for various conditions (acceleration, temperature, and thrust), and the transient behavior. The results of these analyses indicate that the system would operate within very narrow limits of mixture-ratio, chamber-pressure, and center-of-gravity excursions (assuming no propellant sloshing) and would be capable of rapid thrust transients.

Several layouts of the system have been made. One version, assembled from existing ALPS component parts, is shown as a composite photograph in Fig. 13.

Since a complete system has not been available, tests have been limited to subsystems. Feasibility of the pressurization-circuit concept was successfully demonstrated in early 1961. A throttlable gas generator fed by a mock-up of the ALPS generant feed circuit was used to pressurize a heavy-weight test tank from which water was allowed to flow at varying rates to simulate the modulated flow of propellants to an engine. Similar tests of the latest ALPS hardware are scheduled for spring 1965.

C. A Simplified Monopropellant-System Concept

The ALPS gas generator feed circuit has been tested separately as the propellant system for a simplified monopropellant rocket. As is apparent in Fig. 12, this circuit works on a modified "blow-down" principle. This scheme was selected to avoid the use of high pressures, gas on–off valves, and gas regulators, which are necessary in the conventional regulated-gas pressurization systems. In the blow-down operation, the generant tank is partially filled with liquid, then pressurized with gas. As the liquid is withdrawn, no additional gas is pumped into the tank; instead, the pressure is allowed to decay as the gas in the tank expands. This decline in generant feed pressure has no effect on the ALPS propellant-tank pressure, since the generant controller meters the flow of hydrazine to the gas generator so as to maintain a constant pressure in the propellant tank.

It was recognized at the inception of the ALPS program that this circuit could be substituted for the more conventional regulated-gas-pressurized feed system used in the *Ranger* and *Mariner* midcourse propulsion units. To convert it to this use, it was necessary only to connect the generant controller (regulator) so that it sensed its own outlet pressure, since this would cause the controller to deliver a constant outlet pressure. Such a system was built and tested. When fitted with a *Ranger* 50-lb-thrust engine, the generant controller delivered hydrazine to the injector at a steady-state pressure which varied no more than $\pm 1/2$ psi for the rated firing duration of 190 sec. The system operated smoothly and stably.

If this blow-down feed circuit were combined with an engine utilizing a room-temperature spontaneous catalyst, the system would be substantially simpler than the type of monopropellant propulsion system which has been used on board the *Ranger* and *Mariner* spacecraft. Figure 14 shows schematic diagrams of the *Mariner IV* system and a possible successor based on the scheme outlined above. It is also apparent that the latter system could be made throttlable by installation of a motor-operator to vary the compression of the spring in the liquid regulator, since the setting of this spring determines the propellant pressure delivered to the injector. This system may also provide a starting point in the development of heat-sterilizable propulsion systems. Substitution of a metal expulsion diaphragm for the present rubber bladder and use of a valve with metal seals would eliminate all the parts that are obviously susceptible to damage from the currently specified sterilization temperature of 300°F. Testing would probably reveal that further detailed changes are necessary; however, the basic concept seems to be amenable to the conversion.

D. Development of Components for the ALPS and Simplified Monopropellant Systems

Although the complete ALPS system and the simplified monopropellant system remain merely concepts at this time, considerable progress has been made with the component parts. Some of the individual developments that are of intrinsic interest are briefly described below.

1. Engines Burning Earth-Storable Propellants

Five integrated parts comprise the thrust-chamber assembly in the ALPS system: the nozzle, the combustion chamber, the injector, and the two propellant valves. A conventional de Laval nozzle with a bell-shaped expansion section was selected. The contour of the combustion chamber and nozzle were chosen, primarily, to optimize the stress levels for the favored material, pyrolytic graphite.

Thrust chambers constructed of pyrolytic graphite (PG) and its alloys were of special interest because of the desire to avoid regenerative cooling and to maintain a stable geometry (i.e., to minimize throat erosion). The extremely light weight (2.1 g/cc) and excellent high-temperature strength (yield stress = 12,000 psi at temperatures to 4000°F) combine to make these materials attractive candidates for applications where only a slight amount of erosion can be tolerated over very long burning durations. Tests of free-standing chambers have been

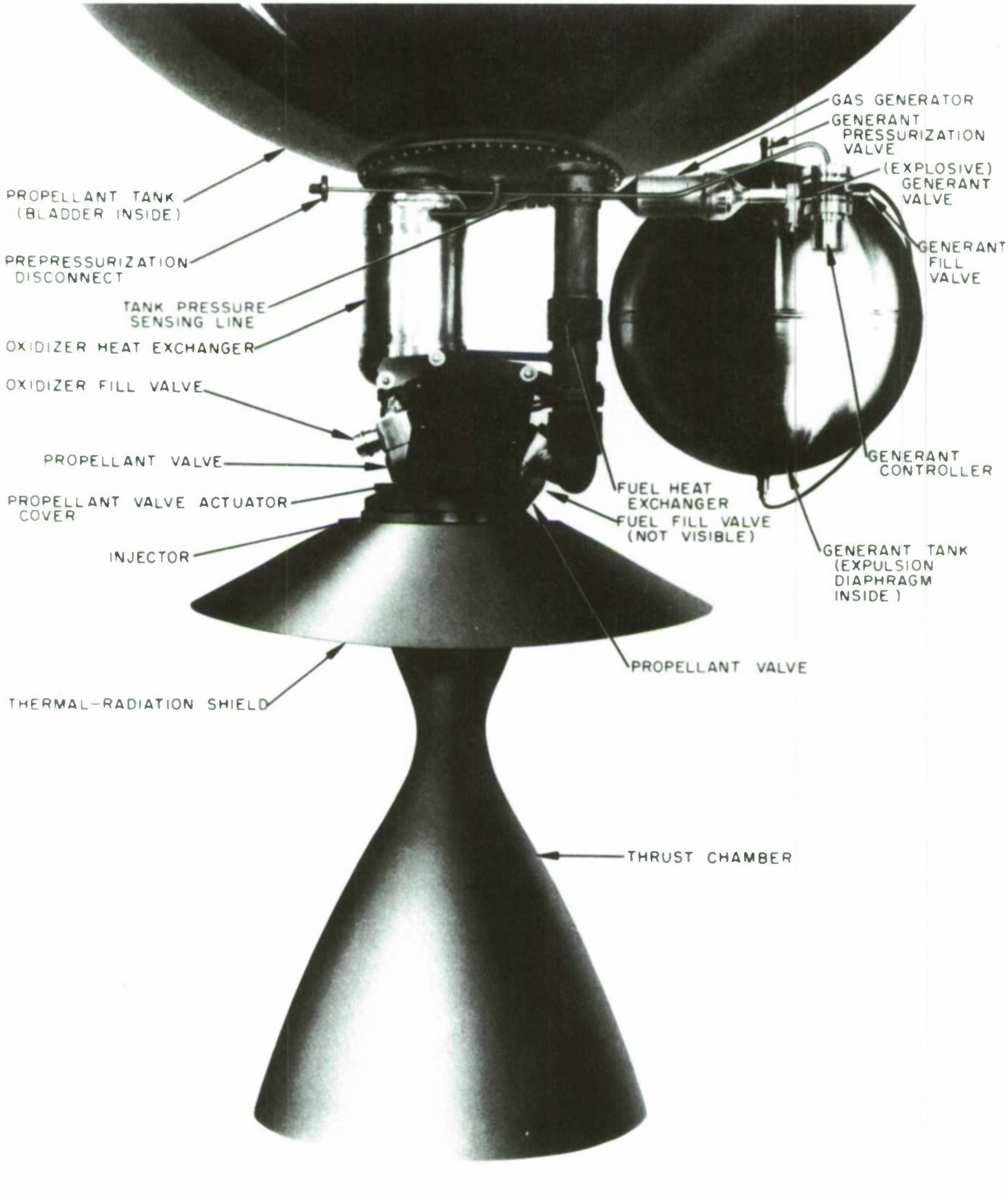


Fig. 13. One possible configuration of the ALPS system, with structural members omitted

REGULATED-GAS-PRESSURE-FED CONSTANT-THRUST SPACECRAFT PROPULSION SYSTEM, MONOPROPELLANT HYDRAZINE (*MARINER* IV TYPE, 1964)

REGULATED-LIQUID-PRESSURE-FED CONSTANT-THRUST SPACECRAFT PROPULSION SYSTEM, MONOPROPELLANT HYDRAZINE (ADVANCED *MARINER* TYPE, 1966-1969

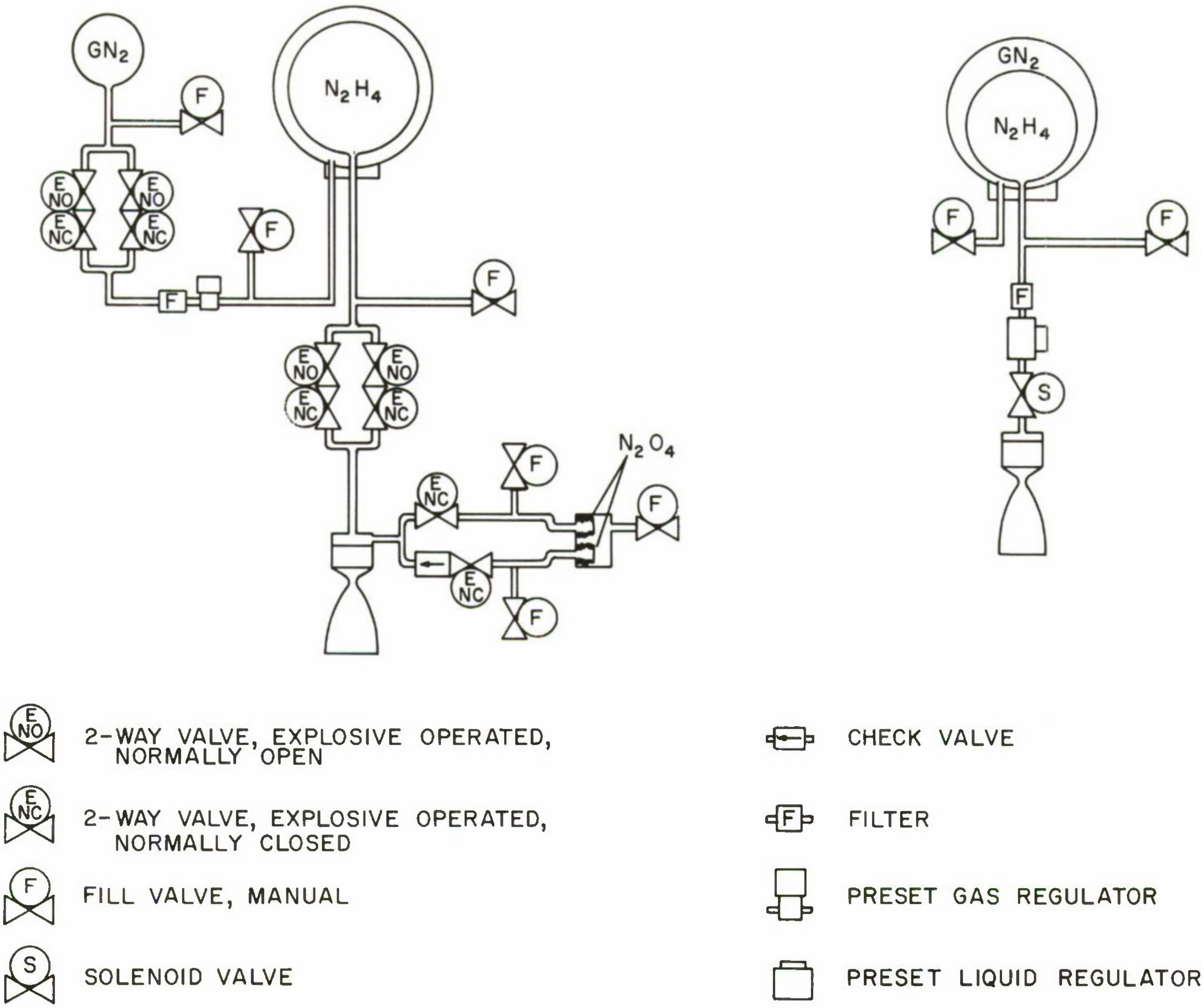


Fig. 14. *Mariner IV* monopropellant propulsion system, compared with simplified configuration made possible by use of a blow-down feed system and a spontaneous hydrazine catalyst

conducted at the 100-lb and 2000-lb thrust levels, with durations of the order of several minutes. Figure 15 shows a chamber of this type in operation during a test at the 100-lb thrust level. (The image at the lower right resulted from a mirror placed behind the chamber.) At present, the outstanding problem with these free-standing structures is the difficulty of attaching them to injectors. A braze-development program now in progress shows promise of solving this problem in the near future.

Several composite chamber designs (i.e., units constructed of two or more materials) are also being evaluated. Composite chambers possess two attributes: the best features of several different materials can be

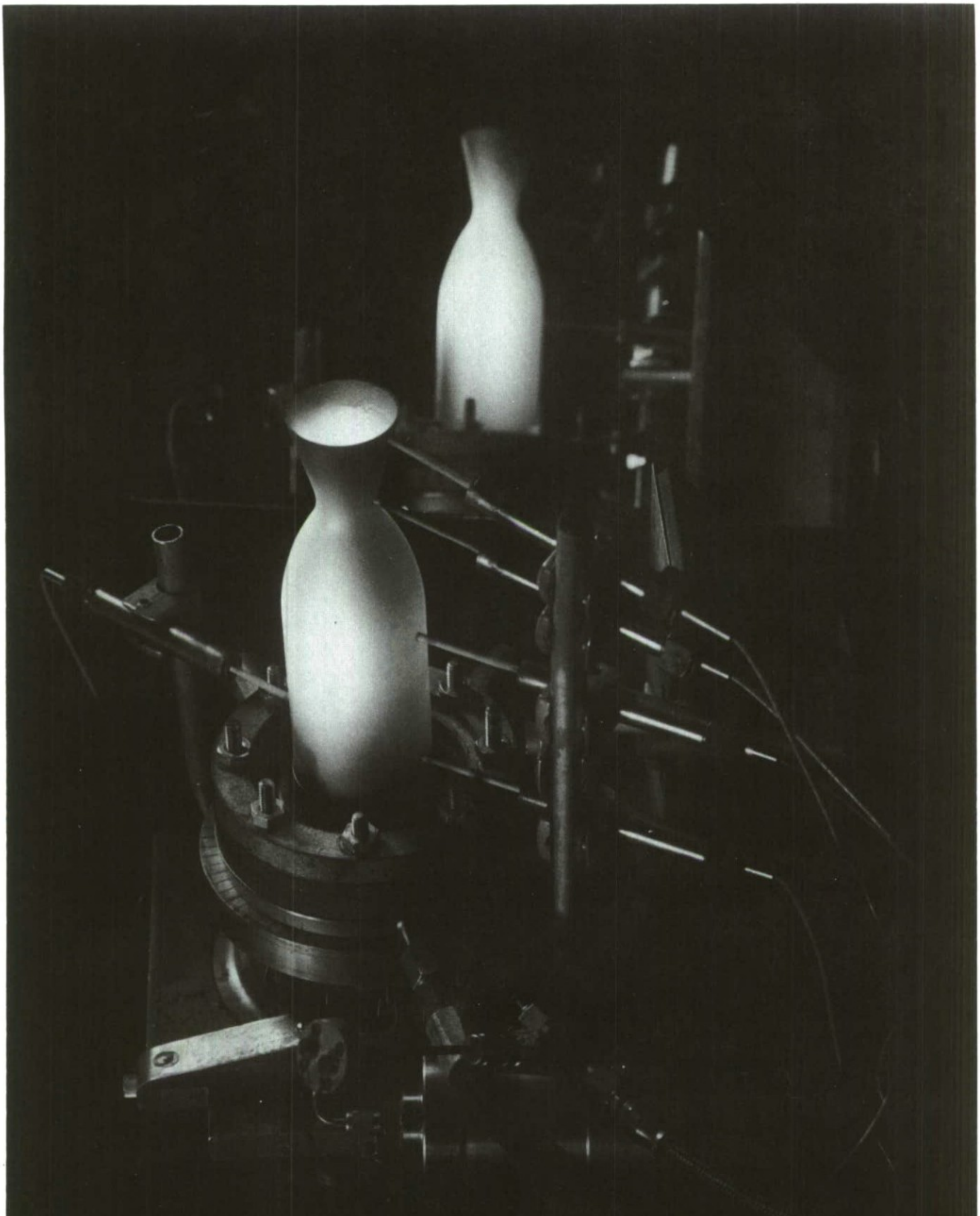


Fig. 15. Free-standing pyrolytic graphite thrust chamber in operation

exploited, and the outer-wall temperature can be more easily controlled. The latter may be very important where a "buried" installation is desired. In addition to PG, such materials as silicon carbide, Refrasil, molded carbon, etc., are used in these chambers. No matter what materials are chosen, however, success is clearly dependent on the injector design, since it controls erosion and heat-transfer rates, as discussed in Section II of this Report.

The prime objective of the ALPS injector development program, therefore, has been to achieve a simple, reliable design which is compatible with the chamber wall. Also, it is to be throttlable over a 10:1 range, and restartable. The initial design consisted of a single impinging-jet-doublet element producing 2000 lb of thrust when burning nitrogen tetroxide (N_2O_4) and hydrazine (N_2H_4) at a chamber pressure of 150 psia. Throttling was accomplished by the movement of a conical pintle in each orifice so as to vary the annular flow area between the pintle and orifice. Early in the test program, it became evident that serious combustion problems existed, since only about 67% of theoretical combustion efficiency was being realized at the full-flow condition. The investigation of this problem led to the combustion-effects experiment described in Section II.

After consideration of several alternative injection schemes, it appeared that the concept of thin-sheet impingement was an advantageous method of accomplishing good mixing of these highly reactive propellants. An impinging-sheet-doublet element was chosen in order to take full advantage of the existing knowledge concerning unlike doublets. An advantage of this technique over the use of simple impinging-stream elements is the lower susceptibility to misalignment. It is expected that the "ignition-spike" problem experienced in most storable-propellant injector configurations when started at very low ambient pressures will also be encountered with the thin-sheet element. (This phenomenon is of particular interest in the present program because of the susceptibility of the pyrolytic graphite thrust chambers to damage from such sharp-transient pressure waves; an investigation of this problem is scheduled for the second half of 1965.)

Each sheet of the element is formed by directing a jet against a suitable solid deflector, as shown in Fig. 16. Individual 25-lb-thrust elements of this design have attained reproducible high performance with N_2O_4–N_2H_4 propellants in an uncooled thrust chamber. At this writing, the task of combining a number of these elements in an injector is under way. Throttling will be accomplished by closing off a discrete number of elements with an appropriate valving arrangement.

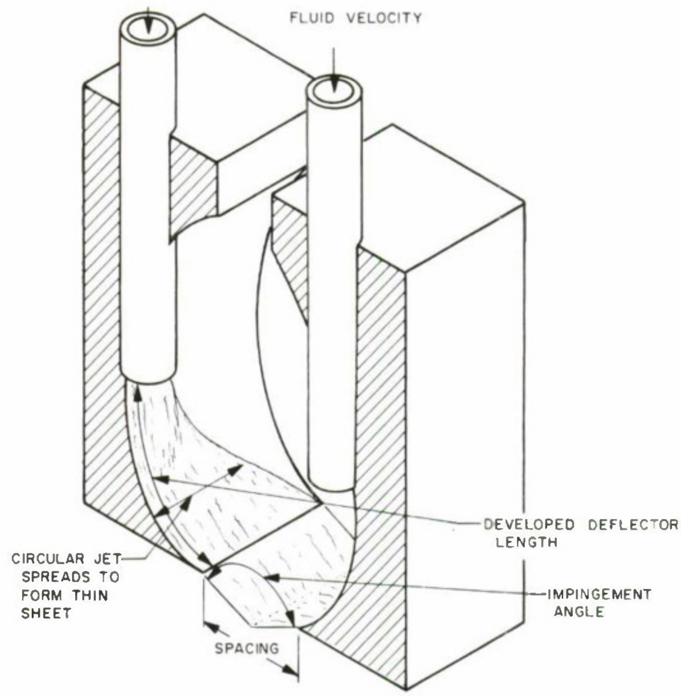


Fig. 16. Four pertinent design parameters for impinging-sheet injector elements

Aside from throttling the propellant flow during firings, the most rigorous requirement for the propellant valving is that it must remain leak-tight for long periods without cold-welding shut. The propellant valves are the only valves in the system which must open after exposure to the hard vacuum of space. An attempt is being made to obtain the required qualities by using a conical ring seal, built up of many thin laminae which are, alternately, of metal and a compliant subliming polymer. The idea is that the edges of the metal discs will seal tightly if given flexible support by the polymer, so that local adjustment can be made for concentricity errors, contamination, local surface defects, and thermal expansion or contraction. Also, it is hoped that the redundancy in the seal provided by the multiplicity of laminae and the filling of the circumferential leak paths by the cold flow of the polymer will be effective in reducing leakage. Cold-welding is to be prevented by the generation of a very slight local gas pressure from the sacrificial sublimation of a portion of the polymer laminae.

A primitive test has been performed to determine the sealing capability of such a composite structure. After several "wearing-in" cycles in a test fixture, a 1¼-in.-diameter laminated seal was leaking about 3×10^{-2} standard cc/sec of gaseous nitrogen when pressurized to 300 psi and loaded with a nominal bearing force; no bubbles were detected during a 10-min observation period after the seal had remained under pressure overnight. Current efforts are directed toward optimizing the seal design and incorporating it into a suitable propellant-valve configuration.

2. Expulsion Devices

One of the most important developments needed for reliable starts under weightless conditions is a method of ensuring positive expulsion of propellants from the tank so that bubble-free liquid is fed to the engine. After a survey of the qualities of available positive-expulsion devices, it was concluded that, on the premise of eventual equal reliability, flexible but impermeable bladders would be the most advantageous.

The expulsion bladders in the propellant tank are the most critical parts in the ALPS system, because a common pressurant source is used without valves to isolate the ullage over the fuel from the ullage over the oxidizer. All ALPS designs to date show both the fuel and oxidizer bladders installed in a single pressure vessel. The use of a single nearly spherical propellant tank may provide several advantages. First, of course, a single tank is often easier to package in a spacecraft than several smaller tanks. Second, the feed lines are shorter, cause less pressure loss, and are likely to require fewer seals than more complicated geometries. Finally, the probability that a micrometeoroid will cause a disabling puncture is lessened by the use of a single spherical tank because the exposed area is less and the wall thickness greater than for several smaller tanks holding the same total volume.

Because of the proximity of the fuel and oxidizer bladders in the tank, the only protection against damage to one bladder caused by leakage from the other is to fabricate both of a material that is resistant to both propellants. Finding a suitable material has proven to be a difficult problem.

Where hydrazine is to be stored alone in a tank, a number of elastomers serve adequately. For example, the generant (hydrazine) in the ALPS system is expelled from its tank by a butyl or ethylene propylene rubber diaphragm. These materials are fairly inert and impermeable to hydrazine. Storing the oxides of nitrogen is another matter, however. In the storage of nitrogen tetroxide, three materials problems are controlling: chemical attack, damage from creases, and permeability. Essentially no elastomers are inert to N_2O_4. Inert polymers, such as Teflon, are permeable. Metals alone seem to be reasonably inert and impermeable to N_2O_4, but metals are easily damaged when tight folding causes creases. Creasing is inevitable unless some form of controlled bladder folding is induced, since most surface shapes will collapse with a multitude of random folds which cause creases. Various phases of these materials problems have been studied by JPL and several of its industrial subcontractors. After the elimination of most candidate materials for one reason or another, only Teflon and some metals remained in contention. Current work is concentrated on composite materials, which are built up of layers of Teflon, in various forms, and metals, either as foils or plated layers.

The hemispherical configuration of the ALPS bladders has made the incorporation of folding control somewhat difficult. A number of bladders constructed of different materials and with different folding tendencies have been tested, but none have been completely successful (Ref. 16).

Because bladder technology is somewhat primitive, it has been necessary to take time to develop new methods and equipment for even the most basic tests. Among these are tests for permeation, crease resistance, expulsion, slosh loading, and long-term storage. Figure 17 shows the JPL crease-test machine devised to subject bladder materials to controlled (i.e., reproducible) creasing. As can be seen in the photograph, the sample strip is folded over two inclined blades, so that a crease is formed at the vertex of the angle formed by the blades. The ends are clamped to cables which are used to impart a reciprocating motion to the sample, so that the point where the crease occurs is repeatedly forced back and forth along a definite path. Materials are rated for relative crease resistance on the basis of the number of cycles completed before a dye penetrant leaks through the crease. Materials tested to date have resisted from 0 to 4000 cycles.

As a back-up development to the expulsion bladder, some work has been done with all-metal convoluted

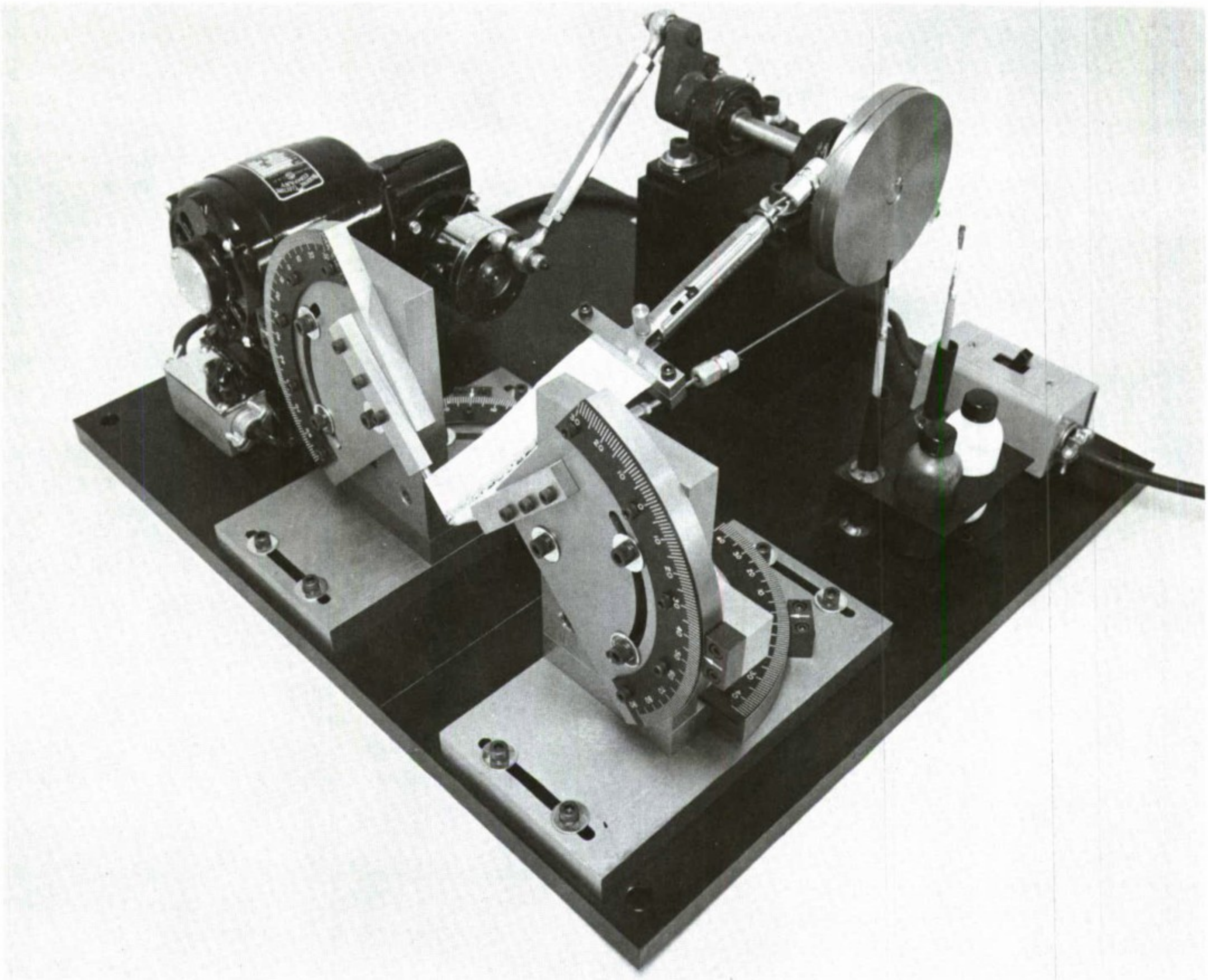


Fig. 17. Machine used in ALPS program to evaluate crease-damage resistance of potential bladder materials

diaphragms. These diaphragms were fabricated by industrial concerns using several different methods. The basic drawback of these devices is that they cannot be recycled. Figure 18 shows a convoluted diaphragm before and after expulsion, and also the typical result of a recycling attempt. Further efforts are being made by JPL subcontractors to achieve recycle capability with specially modified convoluted diaphragms and reversing metal hemispheres. Despite the limitation on cycling, these diaphragms are the only devices tested in this program to date which provide completely impermeable barriers to the propellants and to the pressurizing gas.

3. Tank-Pressurization Components

The propellant tank is pressurized by the decomposition products of hydrazine. Considerable work has been accomplished at JPL in the past with this technology (Ref. 17). A prime reason for using this scheme is that the pressurant is stored as a liquid. When brought in contact with a suitable catalyst, hydrazine smoothly decomposes to a mixture of gases. Although they are somewhat reactive with the oxidizer, the other properties of these gases are advantageous. A low molecular weight (12 g/mole) and freedom from condensation permit a

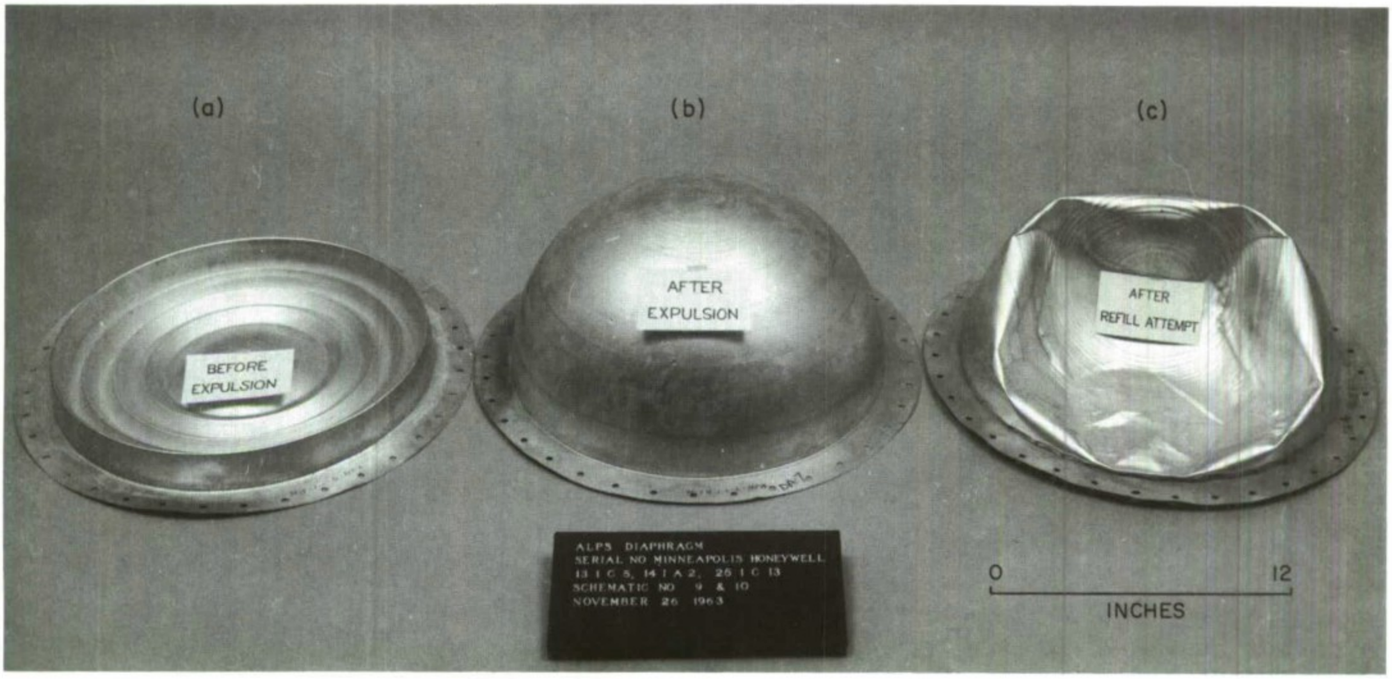


Fig. 18. Life cycle of a convoluted metal diaphragm

(a) Diaphragm properly convoluted and ready for installation. (b) Diaphragm after use in expulsion of fluid from a hemispherical tank. (c) Diaphragm after an attempt to restore the original convolutions by pumping fluid back into the tank.

lightweight system. Gas temperature can be varied by altering the degree of ammonia decomposition, but usually a temperature around 1500°F is selected. This relatively low temperature allows the use of readily fabricated metal alloys for the gas generator and heat exchangers.

The gas generator, in which the hydrazine is decomposed, consists of an injector, an outer shell, and a catalyst bed. Its function is to furnish gas to the propellant tank at a flow rate which is dependent on demand. Three major problems required solution in the development of a suitable gas generator: (1) spontaneous-start capability at ambient temperature, (2) good performance over a wide range of flows, and (3) a safe means of shutoff which prevents heat from soaking from the catalyst bed into the injector and detonating the residual hydrazine trapped there. The availability of Shell 405 catalyst solved the first problem. The second problem is being studied with a number of different design approaches, the most successful of which appears to be a multiorifice, radial-spray injection technique. Commercial poppet-type variable-area spray-nozzle injectors proved to be unstable under certain throttled conditions; hence, new designs are under development which are intended to be insensitive to downstream pressure oscillations. Tests have shown that the third problem is amenable to a straightforward mechanical-engineering design approach, wherein injector cooling is accomplished by heat conduction into either the generant tank or the main propellant feed line.

The generant controller is a remote-sensing liquid regulator which adjusts the flow of generant (hydrazine) to the gas generator so as to keep the propellant-tank pressure constant. To do this, it must vary the generant flow rate over a range of about 20 to 1. The wide range is necessary because the engine is throttlable, and because other vagaries may occur, such as low tank pressures at start which cause momentary flow surges well above steady operating rates. The present design is a simple one-stage spring-loaded pressure-balanced regulator. A ball and a circular seat form the flow-regulating mechanism. The position of the ball is determined by the ratio between the reference force provided by the spring and the sensing-pressure force against a diaphragm. Coil springs having the required combination of high spring force and low spring constant proved to be too big and heavy, so a special Belleville spring assembly was evolved. The assembly allows, through tailoring of the individual springs, any desired characteristic to be obtained. This regulator was used successfully in tests of the simplified monopropellant system mentioned above. It will be given a more rigorous test during pressurization-system investigations scheduled for the near future.

While the ALPS generant controller was under development, the *Ranger* and *Mariner* midcourse systems were suffering from gas-regulator malfunctions. A replacement regulator was designed in which some of the ALPS generant-controller design features were used. It was very simple, but performed extremely well. *Mariner IV* and *Rangers VI* through *IX* were fitted with this regulator, illustrated in Fig. 19. In operation, gas flow is controlled by the clearance between the ball and the seat. When the outlet pressure is below the set point, the force of the Belleville spring assembly is greater than the opposing force caused by the outlet pressure acting on the sensing diaphragm. This force imbalance causes the push-rod to move the ball off the seat. The flexure ring, shown unsectioned, provides support so that the very thin metal diaphragm is not ruptured by the gas pressure.

Among the several types of on–off valves that have been evaluated, the most successful is a very simple manually operated fill valve, which exploits the principle of line contact to achieve maximum seat stress. A screw is used to press a precision ball against a carefully lapped circular seat created by the intersection of a bore with a flat surface. Very low leakage rates are obtained with this design, which was used in the *Mariner IV* system.

Reliable valve sealing depends to some extent on keeping particulate matter out of the system. Filters are used to remove all particles larger than 11 μ from all fluids entering the system and from the generant flowing into the generant controller. These filters consist of stacks of etched discs. The etched pattern on one side of each disc provides a tortuous path from the outer edge to a hole through the center. Such filters are easily made, low in cost, and can be cleaned for reuse. This type of filter was selected for use in ALPS after its successful performance in the *Mariner* systems.

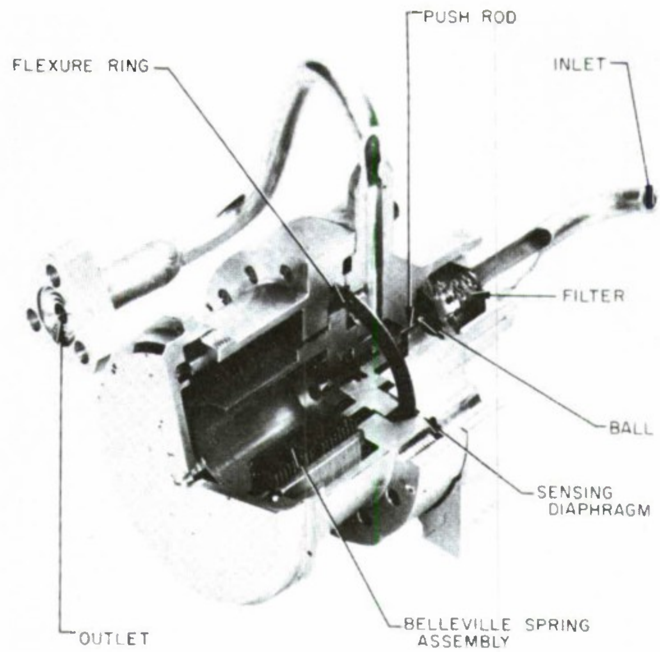


Fig. 19. Important design details of the gas regulator used in the *Ranger* and *Mariner* monopropellant propulsion systems

E. Current Work

Investigations are continuing in the component developments described above, and a number of other component programs are being carried forward. These investigations include studies of heat exchangers, electromechanical valve actuators, and the compatibility of materials with propellants. Details of the overall program are periodically reported in the JPL bimonthly publication, *Space Programs Summary, Vol. IV*.

NOMENCLATURE

A	area, in.^{2}
c^*	characteristic exhaust velocity, ft/sec
d	diameter, in.
G	local flow rate per unit area, lb/(sec in.^{2})
L	distance from impingement point, in.
N	number of elements in injection scheme
$\mathbf{P}$	momentum rate, (lb ft)/sec^2
r	mixture ratio
V	mean velocity, ft/sec
$\dot{w}$	flow rate, lb/sec
$\dot{w}_{el}$	total flow rate of spray $= \dot{w}_{1el} + \dot{w}_{2el}$, lb/sec
α	impingement angle, deg
δ	density of fluid, lb/ft^3
η_c	combustion efficiency
η_m	uniformity criterion
θ	angle of rotation about resultant momentum line, deg

Subscripts:

B	boundary
el	element
S	side
th	theoretical
$unif$	uniform
1	first component of spray
2	second component of spray

REFERENCES

1. *Voyager Design Study: Vol. III. Subsystem Design*, Document 63SD801, General Electric Co., Missile and Space Division, Valley Forge, Pennsylvania, October 15, 1963.
2. *Voyager Design Studies: Vol. IV. Orbiter-Bus Design*, TR 63-34, AVCO Corp., Research and Advanced Development Division, Lawrence, Massachusetts, October 15, 1963.
3. *Investigations of Space Storable Propellants*, RMD Report 6028-F, Thiokol Chemical Corp., Reaction Motors Division, Denville, New Jersey, January 1964.
4. Jaffe, L. D., "Effects of Space Environment Upon Plastics and Elastomers," *Chemical Engineering Progress Symposium Series*, Vol. 59, No. 40, 1963.
5. Wiederhorn, N., *The Space Environment and Its Interactions with Liquid Propellants and Their Storage Systems*, Report No. 63270-02-01, Arthur D. Little, Inc., Cambridge, Massachusetts, September 1961.
6. Whipple, F. L., "On Meteoroids and Penetration," *Journal of Geophysical Research*, Vol. 68, No. 17, September 1963, pp. 4929–4939.

REFERENCES (Cont'd)

7. Rupe, J. H., *The Liquid Phase Mixing of a Pair of Impinging Streams*, Progress Report No. 20-195, Jet Propulsion Laboratory, Pasadena, California, August 6, 1953.

8. Rupe, J. H., *A Correlation Between the Dynamic Properties of a Pair of Impinging Streams and the Uniformity of Mixture-Ratio Distribution in the Resulting Spray*, Progress Report No. 20-209, Jet Propulsion Laboratory, Pasadena, California, March 28, 1956.

9. Rupe, J. H., "A Technique for the Investigation of Spray Characteristics of Constant Flow Nozzles," *Third Symposium on Combustion, Flame and Explosion Phenomena*, Madison, Wisconsin, 1948. The Williams & Wilkins Co., Baltimore, Maryland, 1949, pp. 680–694.

10. Rupe, J. H., *On the Dynamic Characteristics of Free-Liquid Jets and a Partial Correlation with Orifice Geometry*, Technical Report No. 32-207, Jet Propulsion Laboratory, Pasadena, California, January 15, 1962.

11. Elverum, G. W., Jr., and Morey, T. F., *Criteria for Optimum Mixture-Ratio Distribution Using Several Types of Impinging-Stream Injector Elements*, Memorandum No. 30-5, Jet Propulsion Laboratory, Pasadena, California, February 25, 1959.

12. Rupe, J. H., *An Experimental Correlation of the Nonreactive Properties of Injection Schemes and Combustion Effects in a Liquid-Propellant Rocket Engine: Part I. The Application of Nonreactive-Spray Properties to Rocket-Motor Injector Design*, Technical Report No. 32-255, Jet Propulsion Laboratory, Pasadena, California, July 15, 1965.

13. Rowley, R. W., *An Experimental Investigation of Uncooled Thrust Chamber Materials for Use in Storable Liquid Propellant Rocket Engines*, Technical Report No. 32-561, Jet Propulsion Laboratory, Pasadena, California, February 15, 1964.

14. Elverum, G. W., Jr., and Staudhammer, P., *The Effect of Rapid Liquid-Phase Reactions on Injector Design and Combustion in Rocket Motors*, Progress Report No. 30-4, Jet Propulsion Laboratory, Pasadena, California, August 25, 1959.

15. Johnson, B. H., *An Experimental Investigation of Combustion Effects on the Mixing of Highly Reactive Liquid Propellants*, Technical Report No. 32-689, Jet Propulsion Laboratory, Pasadena, California, July 15, 1965.

16. Porter, R. N., and Stanford, H. B., "Propellant Expulsion in Unmanned Spacecraft," Paper 868B, presented at the *SAE-ASME Air Transport and Space Meeting*, New York, April 27 to 30, 1964.

17. Lee, D. H., and Evans, D. D., *The Development of a Heated-Hybrid Generated Gas Pressurization System for Propellant Tanks*, Technical Report No. 32-375, Jet Propulsion Laboratory, Pasadena, California, February 15, 1963.